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JUN 79 D E LAUTNER , A J MAREK , J R PERKINS F33615-78-C-2018

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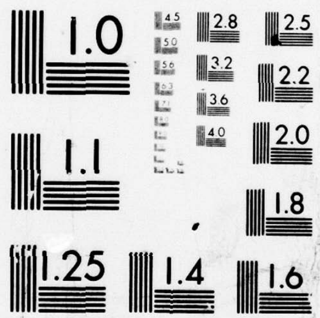

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POWER SYSTEM CONTROL STUDY
PHASE I - INTEGRATED CONTROL TECHNIQUES

VOUGHT CORPORATION
AN LTV COMPANY
DALLAS, TEXAS

OCTOBER 1979

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INTERIM REPORT FOR PERIOD 15 JUNE 1978 – 15 JULY 1979

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This report documents the Phase I results of a two phase study program which addresses the integration of advanced power system control technologies into a reliable and fault tolerant system. The advanced control technologies integrated include electric engine start, automatic load management, microprocessor control implementations, design techniques for providing "no-gap" power and an all "solid state" electric power distribution system. Electric system performance requirements are → over

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established and preliminary designs of an integrated "baseline" control system for single and multiengine aircraft for the 1990 operational time period are presented. Finally, stability analysis requirements and procedures are established for each of the two designs. During Phase II, detail design of the power control system will be completed and a stability analysis will be conducted.

FOREWORD

This interim technical report describes the effort conducted by Vought Corporation under Contract F33615-78-C-2018. The effort was sponsored by the Air Force Aero Propulsion Laboratory, Air Force Wright Aeronautical Laboratories, Wright-Patterson Air Force Base, Ohio.

The work reported herein was performed during the period 1 July 1978 to 15 June 1979 under the technical direction of J. R. Perkins (Vought Corporation), Project Engineer. The report was prepared by A. J. Marek and D. E. Lautner.

Sundstrand, General Electric and Westinghouse were retained under subcontract to provide pertinent data, information and consultations on the program primarily in the areas of power generation and control.

This report covers phase 1 of a planned two phase program concerning power system control technology. Phase I scope was directed toward evolving power system configurations for single and multi-engine aircraft for the 1980 to 1990 time period. Phase II will address developing detail designs, preparing analytical models and performing stability analysis on the single and multi-engine systems.

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SECTION I
INTRODUCTION

1.1 BACKGROUND

As aircraft capabilities continue to expand and the cost of manned aircraft systems rapidly escalate, the electrical system is called upon to provide higher quality and more reliable power at lower cost. This has arisen mostly from avionic equipment sophistication which requires uninterrupted power for many flight critical loads as well as many mission completion loads. Additionally, the avionic equipment sophistication requires controls which tax the capabilities of crew members. To alleviate the crew workload problem, the trend has been toward automation of control functions. Consequently, the electrical system has become highly advanced and complex with the use of solid state switching components in conjunction with computer control technology. A computer controlled solid state distribution system known as EMUX (Electrical Multiplexing) has been developed and implemented on a large aircraft (B-1). This system does not, however, control the generators or main bus contactors. Integration of these control functions can become complex and the total system may have instabilities which could lead to failures and even total loss of electrical power if not applied correctly.

Solid state power controllers have been developed and this technology is now considered off-the-shelf. However, additional efforts are being made to improve performance and to lower the life cycle cost of these components.

Advanced electrical power generator/starter systems have been developed. One starter/generator concept utilizes rare earth permanent magnet solid rotors along with solid state electronics to generate three-phase 400 Hz power and provide engine start capability. Another concept known as Integrated Starter Drive (ISD) utilizes a modified CSD configuration to provide a hydrostatic drive capability.

Other advanced technologies include microprocessor implementation of generator control functions, automatic load management via EMUX, hybrid bus controllers and static inverters which enhance the capability for providing an uninterrupted power bus.

1.2 PROGRAM OBJECTIVES

The objective of this program phase was to evolve a reliable, fault tolerant system through the integration of advanced generator control, regulation and protection functions; and application of advanced power distribution, bus control and protection techniques. More specifically stated, the objective was to evolve and apply technology for a cost effective generator control system which will not sustain cascading type faults that might lead to complete loss of electrical power. Emphasis was placed on utilizing advanced technologies such as the electrical multiplex system (EMUX) for control functions, the permanent magnet VSCF and advanced IDG starter/generators, microprocessor control of generator and buses, and solid state power control and protection components.

1.3 METHOD OF APPROACH

The study is divided into two phases. In general, during Phase I various integrated control techniques which have application in aircraft electrical power systems were studied, and an assessment of their stability and cost effectiveness was made. Electrical control system performance requirements were established, design philosophies for the application of these systems in advanced technology aircraft were established, and a preliminary design of an integrated control system was made. During Phase II, a detailed design of an electrical system for a single engine and multi-engine aircraft will be made. Finally, a stability analysis will be conducted on each of the two designs to verify their performance capability.

More specifically, the method of approach to the power system control study during Phase I was as follows:

- Conduct technology survey - This included a review of existing reports specifications, etc., and to a greater extent a review of data furnished by manufacturers of advanced technology electric systems.
- Establish Electric System Requirements - Performance requirements were established for a single engine and a multi-engine aircraft. These requirements are based on known and projected missions for the 1990 operational time period.
- Conduct trade studies - Various trade studies were conducted. These include power generating concepts (VSCF, CFG, IDG, CSD), generator control concept (Microprocessor, EMUX), electric engine start concepts, EMUX processor redundancy requirements and power bus arrangements.

- o <u>Develop Preliminary Designs</u> - A preliminary design was developed for a single engine and a multi-engine (4) electrical systems. The designs were based on results of the trade studies, data supplied by electric power system manufacturers and evaluated control concepts which were determined to be the best for meeting the established system requirements.

SECTION II

SUMMARY

In this study, advanced system concepts were examined. Such concepts include the use of (1) solid state switching in areas of power generation and control, bus and load switching, and signal sensing; (2) engine electric start implementation; (3) smart terminal (microprocessor controllers) applications in the areas of the GCU, Universal Multiplex Terminal, and integrated load management center designs; (4) dynamic load management and generator control via the EMUX sensory and data processing capabilities; (5) distributed load management; (6) automated and continuous Built-in-Test; and (7) MIL-STD-1553 compatible multiplexing using TSP (Twisted-Shielded Pair) and F/O (Fiber Optic) data busing techniques. In addition, fundamental electrical system considerations as governed by MIL-STD-704 and AFSC DH 2-3 were examined. Emphasis was placed on utilizing new technologies in deriving an advanced electrical system which provides improved performance and higher quality power primarily in the areas of reduced voltage transients and power interruptions, than is presently achievable in conventional systems as defined by MIL-STD-704.

Two power generating concepts (cycloconverter VSCF and IDG) were determined to be the best for the 1980-1990 time period electric systems. Electric engine start provides advantages over the more conventional self start concepts under certain conditions. The EMUX controlled power distribution system offers definite advantages over the conventionally implemented distribution system. Providing a true "gapless power" AC bus is not possible although the power interruption time

can be significantly reduced over that allowed by MIL-STD-704. Solid state and hybrid bus controllers are practical for certain applications.

SECTION III

ELECTRIC SYSTEM REQUIREMENTS

This section establishes the electric system performance requirements for a single and a multi-engine electric system. The requirements establish a framework or baseline for subsequent analysis of control concepts and candidate systems. The rationale was to first establish system requirements and then update or modify the requirements for a specific mission. It is intended that this modification can be implemented without changing the basic power system control philosophy established for either the single engine or the multi-engine aircraft electric system.

3.1 SINGLE ENGINE AIRCRAFT

The single engine aircraft in the 1990 time frame is visualized as a multi-mission vehicle which can be reconfigured to meet a mission of reconnaissance, ground support, electronic-warfare, or fighter. The basic equipment complement includes data processors, navigation and flight controls (fly-by-wire), actuators, communication and weapon systems. The approximate power source requirement is 60 KVA. Performance requirements established for the electrical system are categorized as follows:

1. No single failure shall cause the loss of all electric power or be hazardous to flight safety.
2. The electric system shall provide sufficient power to permit safe recovery of the aircraft with the main power source inoperative.

3. The auxiliary power source shall operate independent of the engine.
4. Physical and electrical isolation shall be maintained between major power management centers.
5. No single feeder fault shall cause the complete loss of power to any load management center (LMC).
6. A "gapless" power bus with sufficient capacity to supply power to loads which are sensitive to power interruptions shall be provided (power interruptions, if present, shall not exceed 20 milliseconds).
7. The electric system shall provide sufficient continuous ground power independent of the engines to operate all maintenance and/or ground related loads.
8. An engine self-start capability shall be provided.
9. The electric system shall contain provisions for the automatic application and removal of loads in groups not exceeding 25 percent of main power source rating.
10. The predicted reliability of the electric power system in providing power to the bus and to <u>all</u> utilization equipment required for the safe return of the aircraft shall not be less than 0.9998 and 0.998 respectively.
11. The predicted reliability of the electric power system in providing power to the bus and to <u>all</u> utilization equipment required for completion of a specified mission shall not be less than 0.995 and 0.990 respectively.

3.2 MULTI-ENGINE AIRCRAFT

The multi-engine aircraft is a four engine vehicle with a multi-mission capability for use in the 1990 time period. The basic equipment complement includes data processors, navigation and flight controls (fly-by-wire), actuators, motors, communication equipment and weapon systems. The approximate power source requirement is 90 KVA. The performance requirements are categorized as follows:

1. No single failure shall cause the permanent loss of more than one generating channel nor prevent the aircraft from completing its mission.
2. Any combination of two failures shall not cause complete loss of power.
3. The electric system shall support safe recovery of the aircraft with all main generating channels inoperative. The auxiliary power source shall operate independent of the engines.
4. Physical and electrical isolation shall be maintained between major power management centers.
5. A single feeder fault shall not cause the loss of rated power to any load management center.
6. Any combination of two feeder faults shall not cause the loss of all power to any LMC.
7. The normal mode of operation shall be "synchronized and isolated". Parallel operation capability may be required for specific missions.

8. A "gapless" power bus with capacity sufficient to support continuously those loads which are sensitive to power interruptions shall be provided. (Power interruptions, if present, shall not exceed 20 milliseconds.)
9. The electric system shall provide sufficient continuous ground power independent of the engines to operate all maintenance and/or ground related loads.
10. An engine self-start capability shall be provided.
11. The electric system shall contain provisions for the application and removal of loads in groups not exceeding 25 percent of one generating channel rating.
12. The predicted reliability of the electric power system in providing power to the bus and to all utilization equipment required for the safe return of the aircraft shall not be less than 0.99995 and 0.991 respectively.
13. The predicted reliability of the electric power system in providing power to the bus and to all utilization equipment required for completion of a specified mission shall not be less than 0.9998 and 0.980 respectively.

SECTION IV

PRELIMINARY SYSTEM DESIGN

The following paragraphs define "baseline" electric system concepts for a single engine aircraft and a multi-engine aircraft. These concepts were established to meet the study objectives and proposed system requirements given in Sections I and III, and were derived from trade studies and data supplied by generator system manufacturers. It should be noted that the defined concepts may or may not apply for all aircraft weapon systems. Definition and evaluation of specific weapon system mission and performance requirements will dictate which of the electric system features are to be implemented.

4.1 SINGLE ENGINE AIRCRAFT

The electric system for a single engine aircraft is defined to consist of the following:

1. One main AC generating channel rated for 60 KVA with a microprocessor implemented GCU for improved logic control and self test capability.
2. Engine electric self start capability.
3. One APU rated for total essential AC load, maintenance AC load or engine start load, whichever is greater.
4. One static inverter rated for total "gapless power" load.
5. One battery rated to provide gapless power to all loads requiring no power interruption. The battery powers loads during the time after main generator shutdown and prior to APU start-up.

6. Five load management centers (LMC's), each center supplied power with a main AC (3ϕ) feeder and an auxiliary AC (3ϕ) feeder.
7. Normal mode of operation of inverter is "standby" and synchronized to the main AC bus. Time required to switch to "standby" bus is 20 milliseconds maximum.
8. Solid state load controllers are used to control and protect load power circuits.
9. Control of power to individual loads is by EMUX. The EMUX system consists of the following:
   - o Two Processors
   - o Approximately 11 MUX/DEMUX (Universal) Terminals of 64 channel capacity each.
   - o One Maintenance Panel
   - o One Control Panel
   - o A Redundant Data Bus
10. System contains capability of sequentially removing individual loads in accordance with a preset priority for load management.

A functional schematic of the baseline system for a single engine (one channel) aircraft is shown in Figure 1. The system consists of three major sub-systems which are: power generation, power bus management and power distribution with EMUX control. These are described in the following paragraphs.

4.1.1 POWER GENERATION

The primary power system consists of an advanced technology generating system which employs either the IDG or permanent magnet VSCF (cycloconverter)

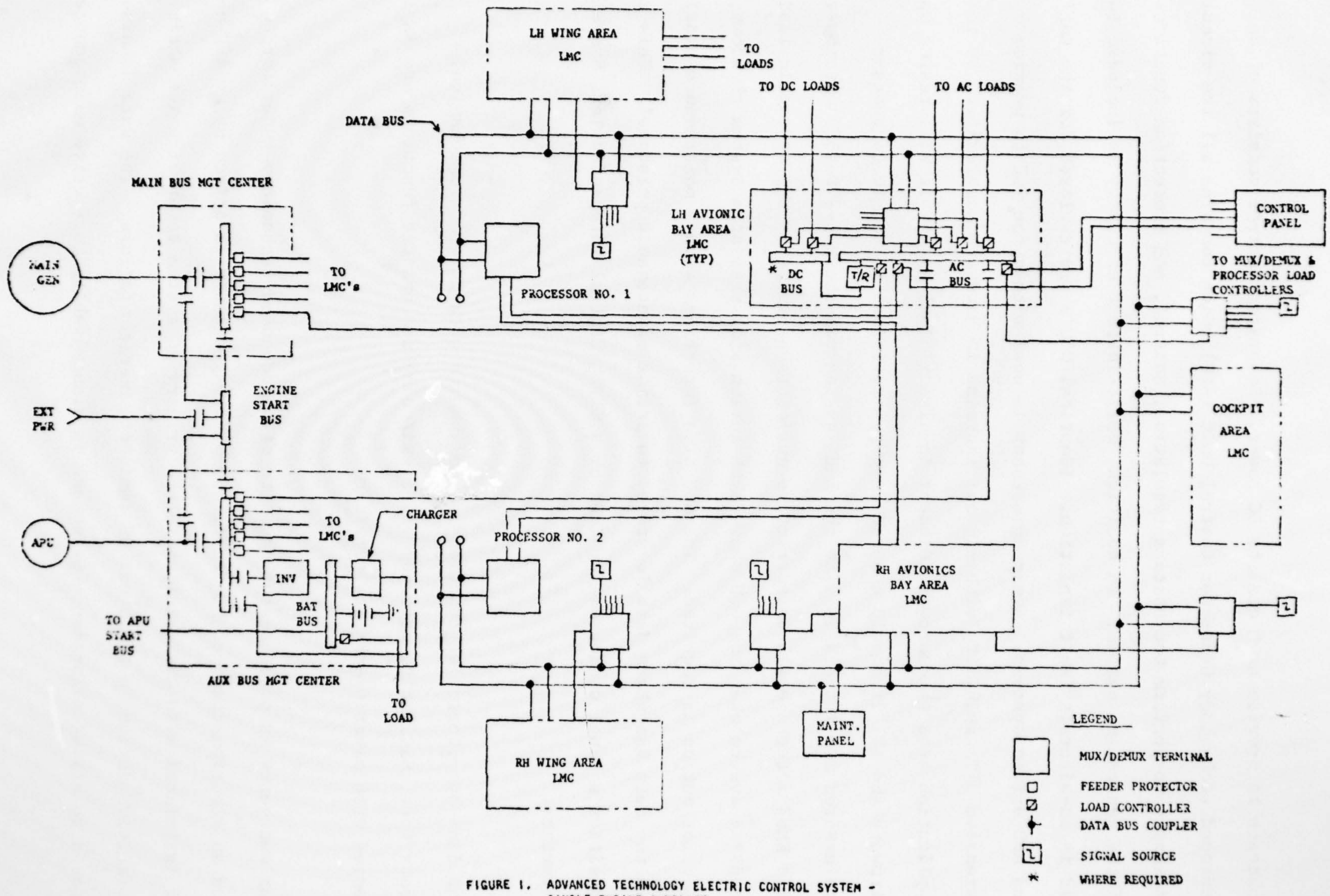

FIGURE 1. ADVANCED TECHNOLOGY ELECTRIC CONTROL SYSTEM - SINGLE ENGINE AIRCRAFT

concepts to provide high quality AC power. Generator control consists of an advanced technology Generator Control Unit (GCU) which contains all the circuitry necessary to perform the voltage regulation, control, and protection functions of the generating system. In addition, built-in-test circuitry is included to aid in on-aircraft fault isolation. Identical GCU's are employed for the engine and APU driven systems. A microprocessor is used within the GCU to provide extensive BIT capability with minimal increase in hardware complexity. In this application area the software element is also minimal as (1) it will reside as firmware and (2) it should undergo very few revisions. The microprocessor implemented GCU interfaces with the EMUX system via the MIL-STD-1553 data bus. This EMUX interface allows corrective action to be taken by the automatic load shedding system when the GCU shows such action will prevent a system shutdown. No pilot action is required. The GCU also interfaces with a maintenance panel via the data bus where data is stored to assist maintenance personnel. This may be either a panel dedicated for EMUX or a CITS (Central Integrated Test System) interface.

The line bus controller function is implemented with an electromechanical contactor. Similarly, external power switching and bus tie functions are implemented with electromechanical devices.

The emergency AC power system consists of an advanced technology generator, GCU and an electromechanical bus controller. The generator is driven by an APU and is interfaced with EMUX similar to the primary AC system to provide automatic corrective action in the event of imminent generator failure. The data is also stored to assist maintenance personnel in accomplishing corrective maintenance

actions. The APU driven generator is rated for total maintenance load, total essential load or engine start load, whichever is greater.

A static inverter in conjunction with a battery is used to supply power to loads in the event of momentary power loss and is rated to supply power to those loads which are sensitive to power interruptions. The inverter operates in a "standby" mode, is synchronized to the primary system and is switched to the bus by a solid state or hybrid bus controller to limit the power interruption time to less than 20 milliseconds. The battery is rated to supply power to the inverter for the time duration required to bring the APU on-line in the event of a main generating system failure.

Main DC power is not provided on a system basis for either the primary or emergency systems. The quantity of DC power being used has diminished with advancements in weapon system designs to the level that an all AC system can be projected for the late 1980's. The only condition remaining in the 1980's which implies a need for DC power is the typical requirement for ground operation of intercoms and communication sets.

In the past, a portion of the aircraft communication system has been powered from a battery bus to permit ground operation independent of the engine, APU or external power cart operation. Operation from this alternate power source was desirable particularly during stores loading or while holding for take-off clearance. This requirement will likely persist into the future.

4.1.2 ENGINE ELECTRIC START

Engine electric start is a viable option with either the VSCF and IDC generating system concepts. In the cycloconverter VSCF system, motor action is accomplished with either a wound rotor or permanent magnet rotor machine. However, it is more difficult with a wound rotor machine since excitation power must be transferred to the rotor even at zero speed. To overcome this problem, a control concept is employed which allows the machine to operate as a wound rotor induction motor in the start mode and as a synchronous machine in the generate mode. This problem is non-existent with the permanent magnet rotor machine (PMG) since it supplies its own excitation. The PMG system can provide start torque with the machine operating either as a synchronous motor or a brushless DC motor. The DC motor equivalent is preferred because of better torque characteristics.

Electric engine start in an IDG system is implemented by initially operating the machine as an induction motor. This is done with very little load on the motor by maintaining the variable displacement pump of the drive at approximately zero stroke. When the motor is near rated speed, motor operation is electrically changed to that of a synchronous motor. The drive hydraulic pump units are controlled by a servo-valve to provide sufficient engine cranking torque needed to overcome the engine inertia. Once ignition occurs, the engine becomes self-sustaining but cranking torque is maintained until starter cut-out speed is reached to minimize the acceleration time. When the engine reaches the underspeed point, the servovalve returns to the generating system mode of governing the speed of rotation and the machine operates as a generator.

4.1.3 POWER BUS MANAGEMENT

Power bus management consists of five load management centers (LMC's), each center being supplied power with a main AC (3Ø) feeder and an auxiliary AC (3Ø) feeder. This feeder redundancy is provided for those loads essential for safe return of aircraft in the event of a feeder fault. Feeder faults are isolated with the combined use of fuses, circuit breakers, RCCB's and electromechanical contactors. Solid state power controllers are not practical for this function because of the relatively high voltage drop and resulting high power dissipation. Hybrid power controllers appear practical but these devices have not been fully developed to date. An overview of the bus management system operation is given in the following paragraphs.

When the aircraft is on the ground without ground power connected and without engine or APU operating, limited power is provided for radio and intercom communications. This power is available from a battery and static inverter under manual control by the pilot. Figure 2A illustrates the bus power flow paths for this condition. A bus controller between the hot battery bus and the booster-inverter is "closed" via a pilot-activated switch. The booster-inverter energizes the standby bus and power is routed to the connected LMC's. Any "normally closed" SSPC's on an energized LMC bus will permit power to flow to the connected utilization equipment (e.g., intercom or radio). This allows operation of limited loads with the EMUX system turned off.

Figure 2B illustrates the engine start sequence. As shown, a pilot controlled RCCB feeds DC power to the APU electric starter. The APU fires up and energizes

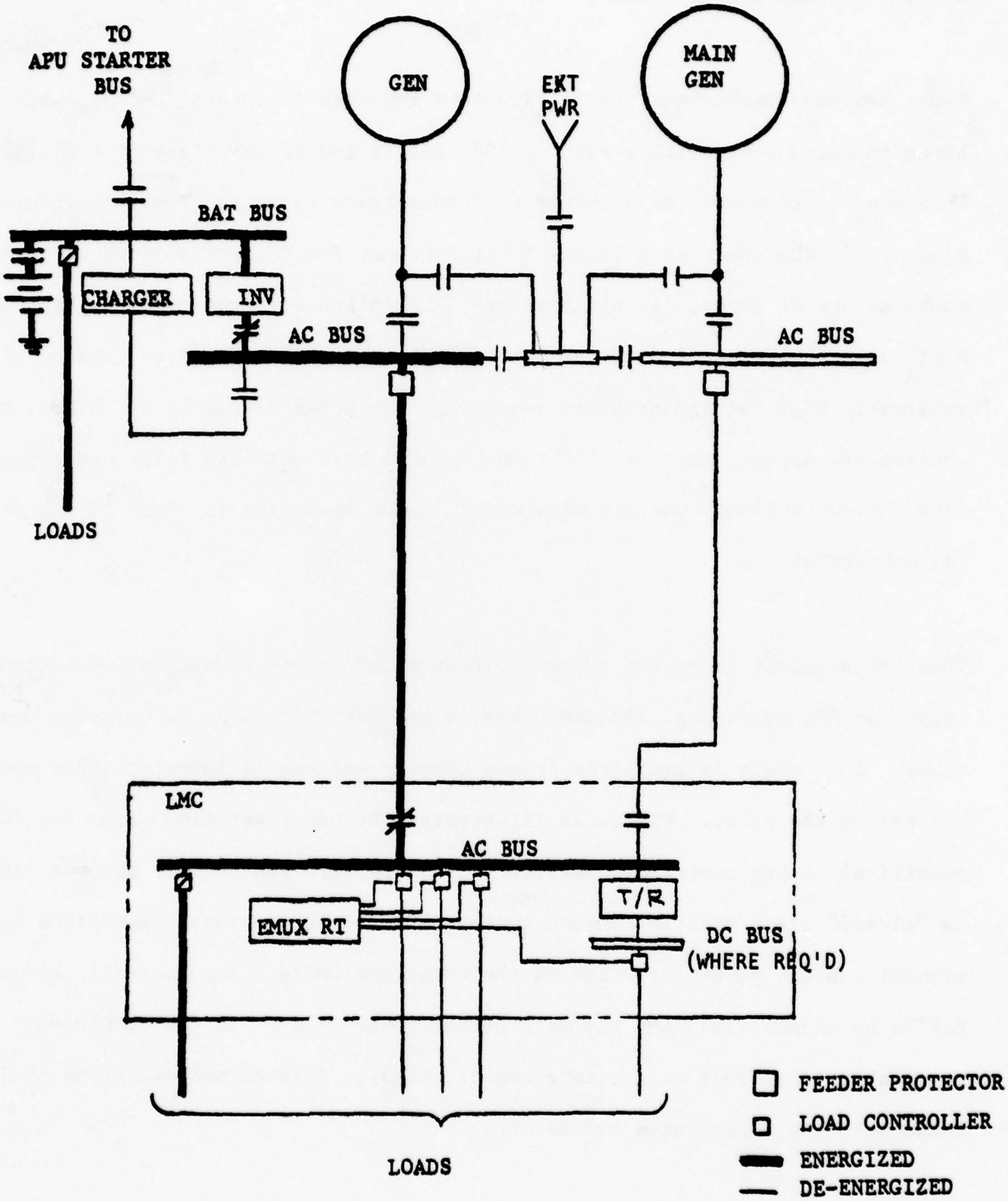

FIGURE 2A GROUND OPERATION - NO GENERATOR, NO EXTERNAL POWER

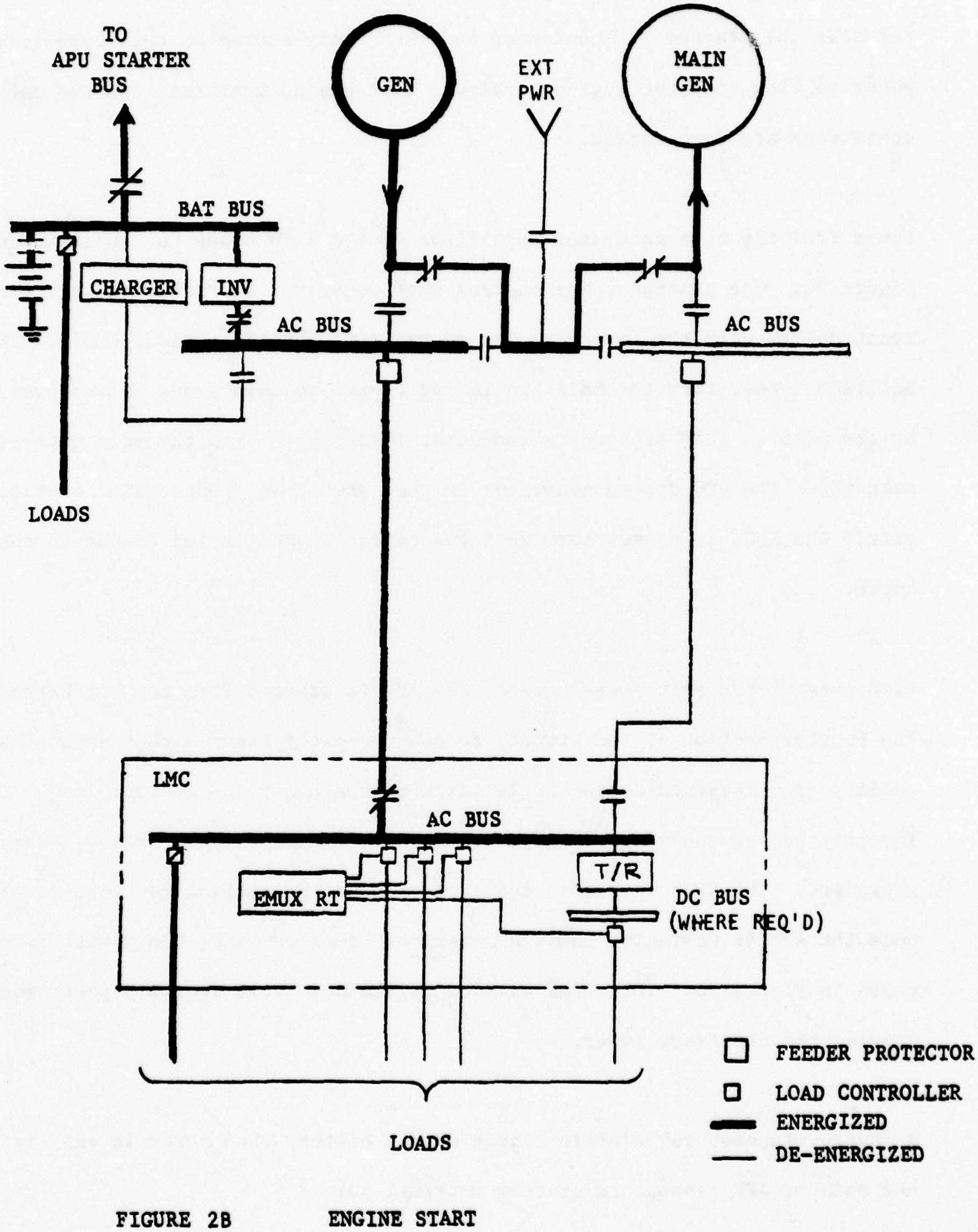

FIGURE 2B ENGINE START

the emergency generator. The APU starter RCCB is opened and the engine crank sequence is initiated with the cockpit throttle control. The throttle control actuates the starter bus contactor and the safety contactor thus permitting power to flow into the engine turbine. With engine ignition, the two safety contactors are deenergized.

Power from the main generator then flows to the main power bus as shown in Figure 2C. The generator/bus control unit regulates the power quality (and synchronizes with the inverter) and routes power to the various LMC's. In addition, power from the main bus is fed to the emergency bus unless overridden by the pilot. This provides a redundant feeder path from the main generator to each LMC. The APU driven generator is then shut down. Once main generator power enters the LMC, it passes through a bus contactor and is fed to the AC three phase buses.

With loss of the main power source, the APU is started from the hot battery bus. The booster section of the standby booster-inverter assures that standby AC power quality is maintained during application of the heavy APU starter load. The inverter powers those essential loads required until the APU driven generator is energized. The EMUX system is available during this period for load control. Once the APU is operating the the generator comes on-line, the power flows as shown in Figure 2D. This flow will be maintained until the main power source can provide the necessary power.

A charger is used to maintain charge on the battery when power is supplied from the main or APU generators or from external power.

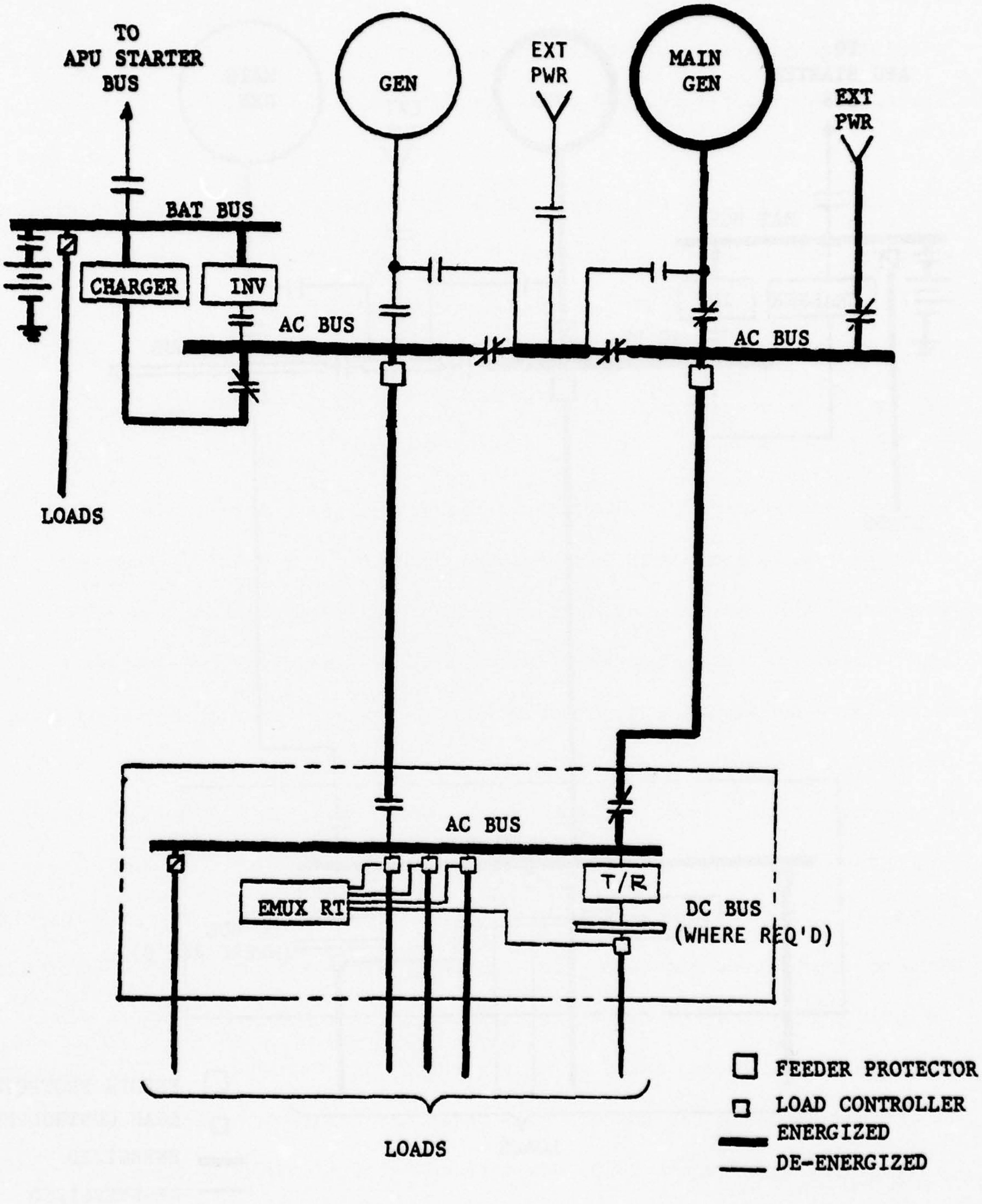

FIGURE 2C NORMAL GENERATOR OR EXT PWR OPERATION

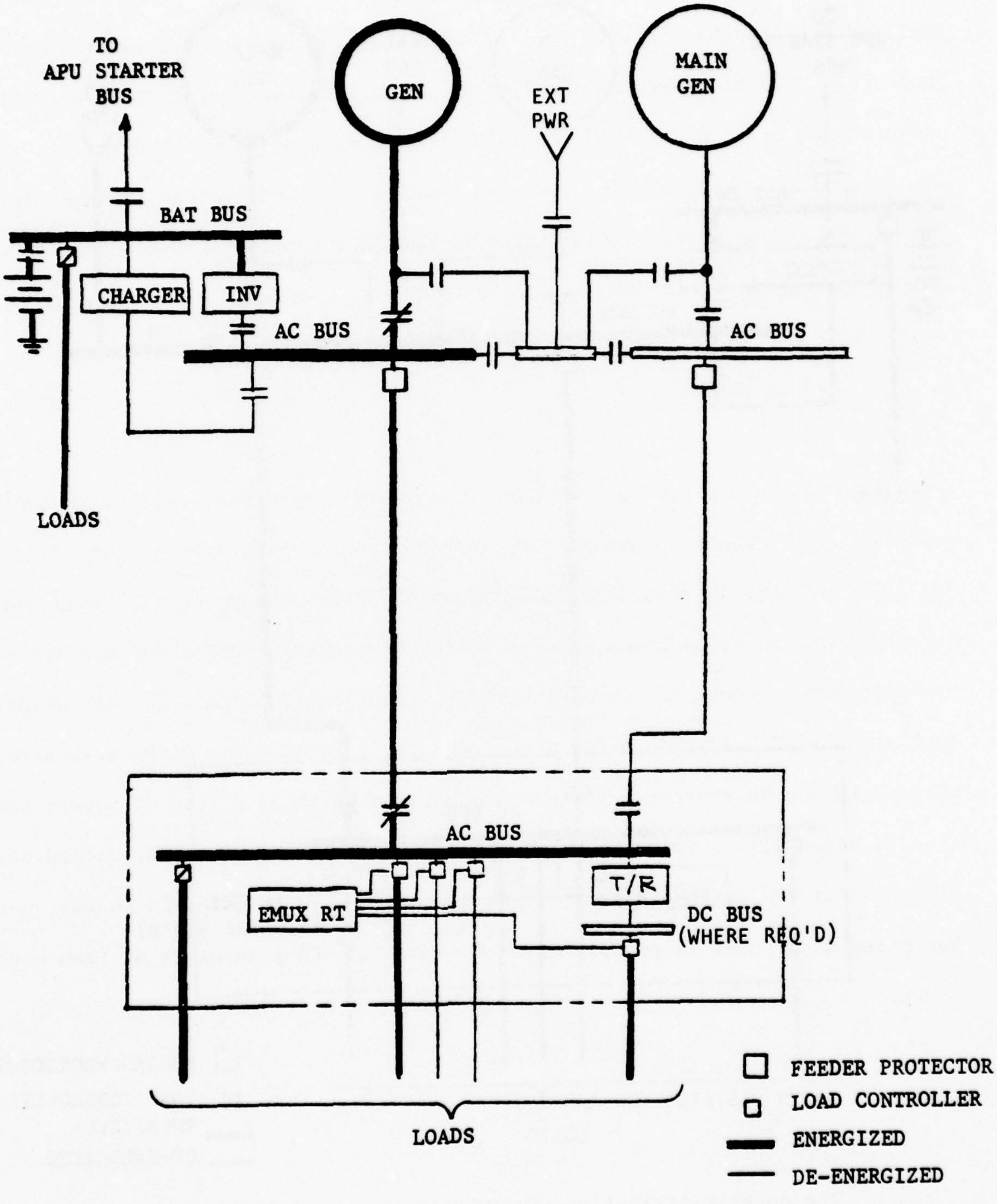

FIGURE 2D EMERGENCY POWER OPERATION

4.1.4 POWER DISTRIBUTION AND CONTROL

The power distribution and control system consists of relatively short load wires, solid state power controllers, signal sources (solid state transducers) and an EMUX control system as shown in Figure 1. Ths signal source provides the input stimulus to the MUX/DEMUX terminal of the EMUX system in response to a mechanical movement (toggle switch action), pressure change, temperature change, etc. Signal adapters are used to convert "black box" signals to be compatible with the input requirements of the MUX/DEMUX terminal. The signal source configuration which offers certain advantages in terms of being all solid-state, not requiring supplemental power and providing a level of built-in-test is a "switched impedance" type as shown in Figure 3. The output circuit is defined in terms of impedance levels. A fixed impedance value is given for a NORMAL ON state and another value is given for a NORMAL OFF state. Fault modes exist when the impedance value is either above or below the normal values. To detect the impedance values, the MUX/DEMUX terminal provides a constant current (10 ma) source that results in voltage levels across the input to the MUX/DEMUX unit which are proportional to the impedance levels.

Power switching between the power bus and the load is performed by solid state load controllers. For a single engine aircraft, approximately 400 controllers are installed in five load management centers. Typical AC load controller distribution by current ratings is given in Table 1.

The built-in-test technique described for the signal source is also included in the controller (see Figure 3). This greatly enhances maintainability and minimizes

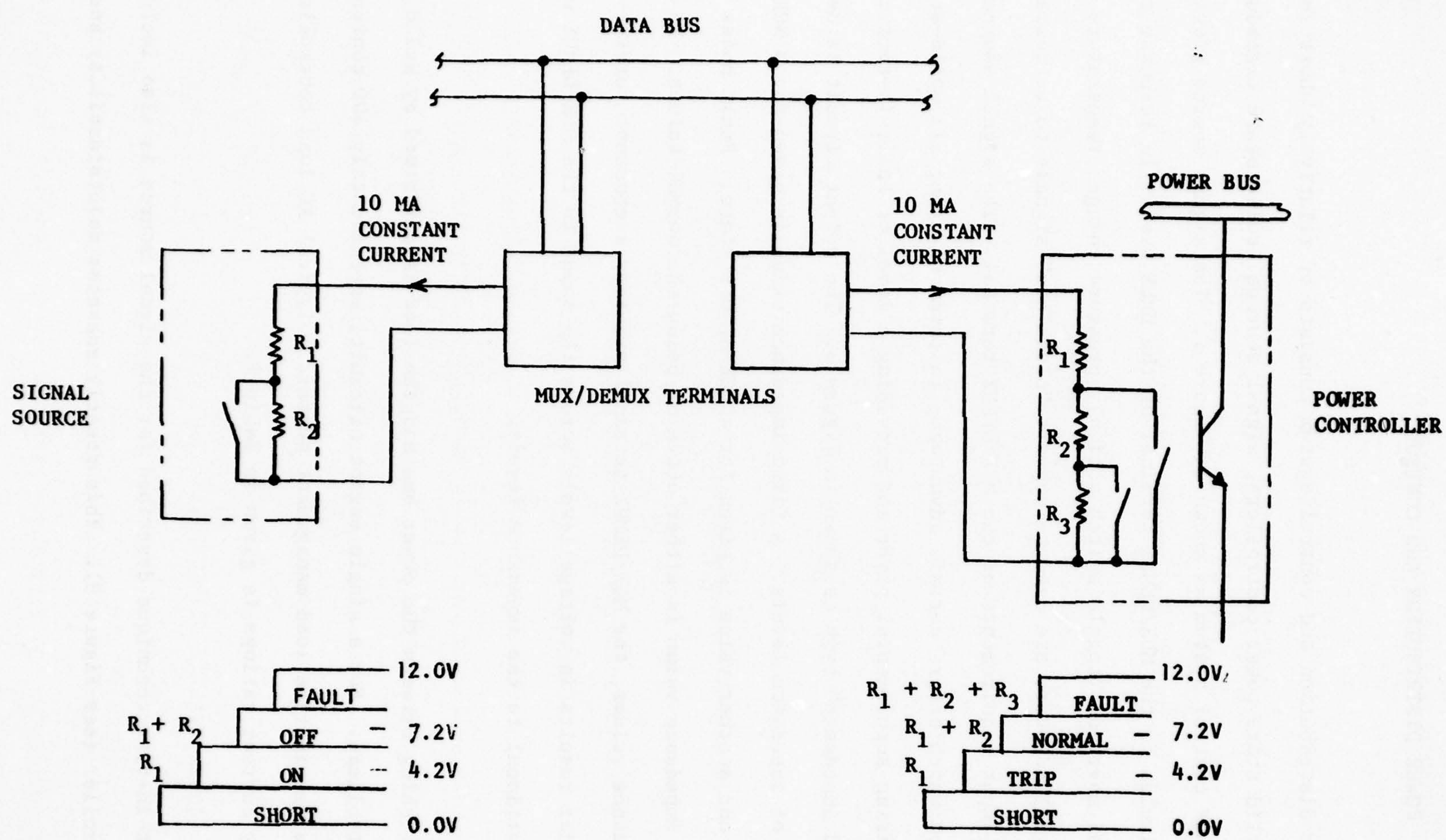

FIGURE 3 CONTROL SIGNAL INTERFACE - SINGLE ENGINE AIRCRAFT

the interconnect wiring needed between the multiplex/demultiplex terminals and controllers. Fault data is provided when the following conditions exist.

- An ON command signal is present and no load current is present.
- An OFF command signal is present .
- An OFF command signal is present and load current is supplied to the load.

TABLE 1

AC LOAD CONTROLLER DISTRIBUTION

LOAD CURRENT	PERCENT OF TOTAL
1/2 A	56
1 A	19
2-1/2 A	14.5
5 A	5.5
10 A	5

This signifies a faulted controller and permits a fault sensing circuit internal to the controller to indicate:

- An open load controller (power switch or fusible link)
- An open circuit between the controller and load
- A dead bus
- A shorted controller
- A faulted control circuit (open or shorted)

All load controllers are installed within load management centers. In addition to the SSPC's, each center contains heat sinks, bus tie switch gear, mounting hardware, connectors, EMUX multiplex/demultiplex (Universal) terminals and an enclosure. The LMC is enclosed to protect against physical damage. The SSPC mounting structure also serves as the heat sink. The LMC is a critical area (somewhat similar to that of the EMUX processor) because it is a focal point where a single "hit" or battle damage can result in significant loss of capability. From this viewpoint, the LMC is even more critical than the processor since it does not have the "complete and separate" level of redundancy. At the same time, it is burdened with the operational and implementation requirements of high heat dissipation and large quantities of signal-power interface points for the distribution and control of power to the many loads within the aircraft. These requirements dictate that the LMC hardware (load controllers, demultiplex terminals and the interconnect-wire system) be very small so that the LMC can be placed behind the avionic equipment. Although placing the LMC behind the utilization equipment is not an essential requirement, it is very desirable because of the survivability criteria. Consequently, the areas designed and usually selected for the LMC's are typically very space limited. For this reason, it has always been a high priority requirement to minimize the size of the LMC equipment. However, the operational and implementation methods used for the load control and protection functions impose limits on the level of miniaturization that can be achieved. These reside primarily in: (1) the number of wire interfaces (signal and power), (2) the intelligence level of the individual controllers, and (3) the power dissipation per controller module combined with the electronic components needed in the controller to achieve its control functions. The power controller intelligence and control requirements are as follows:

- On-Off Power Switch Control
- Circuit Protection
  - -- Current Sense (I^2t function for overload-trip function)
  - -- Trip Control
  - -- Trip Reset Control
  - -- Trip Indication
- Built-In-Test
  - -- Turn-on Delay
  - -- Minimum Current Sense
  - -- Failed Logic
  - -- Fault Indication
- Power-Signal Circuit Isolation

The EMUX system basically consists of hardware similar to that defined for the B-1 airplane. The main difference is the configuration of the multiplexer and demultiplexer terminals. An advanced design uses a common terminal to perform both the multiplex and demultiplex functions, i.e., a "universal terminal". In the universal terminal, any channel can be used for interfacing either a signal source or a power controller. EMUX as generally defined, uses separate terminals to perform the multiplex and demultiplex functions. The common configuration improves design flexibility and reduces logistic support requirements.

The processor, multiplex/demultiplex terminal and maintenance panel hardware and functions are briefly described in the following paragraphs for a single engine aircraft implementation. Also discussed are the Built-in-Test (BIT) capability and criteria or methods for powering-up the EMUX system.

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4.1.4.1 PROCESSOR

Two processors are provided for redundancy, i.e., either processor controls the entire data handling process. One processor operates in stand-by (passively receiving data on the data bus and solving system control equations) and is automatically switched into operation upon failure of the operating unit. Each processor contains a Nondestruct Read Only (NDRO) memory in which are stored all control instructions and the switching equations associated with the control of power to each individual load. The control instructions and equations are programmed into the processor through a software program. Thus, changes in control logic can be accomplished by reprogramming the processor with a paper or magnetic tape, thereby minimizing wiring changes which need to be made in the airplane upon incorporation of a modification. Parity check and monitor circuits within the processor provide a continuous built-in-test capability.

4.1.4.2 MULTIPLEX/DEMULTIPLEX TERMINALS

Typically eleven multiplex/demultiplex universal terminals are located throughout the airplane to pick up control data, and to operate load controllers and special solid state power switches. The multiplex/demultiplex terminal is totally redundant internally and can perform both the multiplex and demultiplex functions. Each terminal has a control capacity of 63 inputs or 63 outputs or combinations of 63 inputs and outputs plus one channel for BIT. The terminals are connected to the processor by a balanced data transmission line. The data line is redundant, consisting of a twisted-shielded pair of wires, and operates in a half-duplex mode at a nominal one megabit rate. The data bus can also be implemented with fiber optics. The multiplex subsystem configuration/operation is in accordance with MIL-STD-1553

4.1.4.3 EMUX MAINTENANCE PROVISIONS

Maintenance provisions on a large airplane will typically consist of CITS (Central Integrated Test System). In absence of CITS, a maintenance panel might be provided; especially for a smaller, single engine airplane. The dedicated type maintenance panel will typically consist of a printer and associated electronics to print BIT information required by maintenance personnel. A simplified maintenance panel provides inhibit-reset and input-output data display. The printer and maintenance panel can be combined into a single unit. All failures detected by the EMUX are printed out by the strip recorder. Each failure is identified by LRU (Line Replaceable Unit) equipment, its location in the aircraft and time of failure. The quantity of data printed is a compromise between the quantity of data required by the maintenance personnel and the ability of the airborne unit to store and print the data in real time.

The displays on the panel consist of a lamp which is illuminated when a channel is inhibited and turned off when the inhibit is reset. A second lamp is provided for data display. Selecting a terminal and channel on the thumbwheel switches causes the lamp to be illuminated if the channel in ON (logic 1). The lamp remains OFF if the channel is a logic "0". A switch is included to test the two lamps. Two switches on the panel are associated with the printer control. The ON-OFF switch controls input power. A three position switch provides for tape advance and tape set.

4.1.4.4 BUILT-IN-TEST (BIT)

BIT is significant capability, although it does not constitute a subsystem or even identifiable added equipment with the EMUX system. The area covered by BIT includes

all of the EMUX, the signal sources and the power controllers including the interconnecting wiring. An important factor contributing to the signal source and power controller BIT is the switched impedance control signal interface illustrated in Figure 3. The BIT includes both circuit functional checks and data monitoring. An example of a circuit test is the power supply voltage tolerance, i.e., an out of tolerance condition results in automatic switch-over to the redundant power supply circuit. Other basic test techniques used with the data handling system are command and response, parity, digital intergration and circulating test bit.

4.1.4.5 EMUX SYSTEM POWER-UP

All the EMUX system components (processors, multiplex/demultiplex terminals maintenance panel) are supplied power from the AC bus. The EMUX system is effectively "hardwired" to the bus and is powered automatically whenever the bus is powered. The primary processor is redundantly powered from AC buses located in separated load management centers to enhance reliability and invulnerability. Similarly, the secondary processor is powered from separate AC buses.

4.1.5 FLY-BY-WIRE SYSTEM

Critical electrical systems, such as fly-by-wire systems, can only be tailored after aircraft system performance characteristics have been defined. Additional levels of redundancy in the areas of power sources, feeders, power distribution circuits, sensory circuits and EMUX hardware to obtain additional levels of reliability and vulnerability must be designed into the basic system. A second

engine driven generator may be required on a single engine aircraft. Zero power interruptions can only be accomplished in DC systems with batteries used as back-up power sources. Power interruptions in AC systems can be reduced as discussed in paragraph (5.1.7) but never completely eliminated upon loss of the primary system.

4.2 MULTI-ENGINE AIRCRAFT

The electric system for a four engine aircraft is defined to consist of the following:

1. Four main AC generators, each rated for 90 KVA.
2. One APU generator rated for 90 KVA.
3. Engine electric start capability.
4. Two static inverters, each rated for total AC "gapless power" load.
5. Two batteries, each rated to provide "gapless" power to all loads requiring no power interruptions. The batteries also power all essential loads during the period after main system shutdown and prior to APU start-up.
6. Nine load management centers (LMC's), each center supplied power with two main AC (3Ø) feeders and one standby AC (3Ø) feeder.
7. The normal mode of operation is split bus with all channels isolated and synchronized. One RH and one LH channel is in operation, each supplying power to one half of the total load. The remaining LH and RH channels are in standby and the APU is OFF.
8. All channels are capable of parallel operation.

9. Normal mode of inverter operation is "standby" and synchronized to the main AC buses. Time required to swtich between buses is 20 milliseconds maximum.

10. System loads are equally divided between bus channels 1 and 3, and 2 and 4.

11. Solid state load controllers are used to control and protect all load power circuits.

12. Control of power to individual loads is by EMUX.

    EMUX system consists of the following:

    - o Four Processors
    - o Approximately 32 MUX/DEMUX Terminals
    - o One Maintenance Panel
    - o One EMUX Control Panel
    - o A Split Dual Data Bus System

13. Control signals are generated by "switched impedance" type signal sources.

14. System contains capability of sequentially removing individual loads in accordance with an established priority for load management.

15. Microprocessor implemented GCU for improved logic control and self-test capability.

A functional schematic of an advanced technology electric control system for a multi-engine aircraft is shown in Figure 4. The system consists of four advanced technology primary generators, an auxiliary power source, provisions for external power, four main power bus management centers and two standby bus managenent centers.

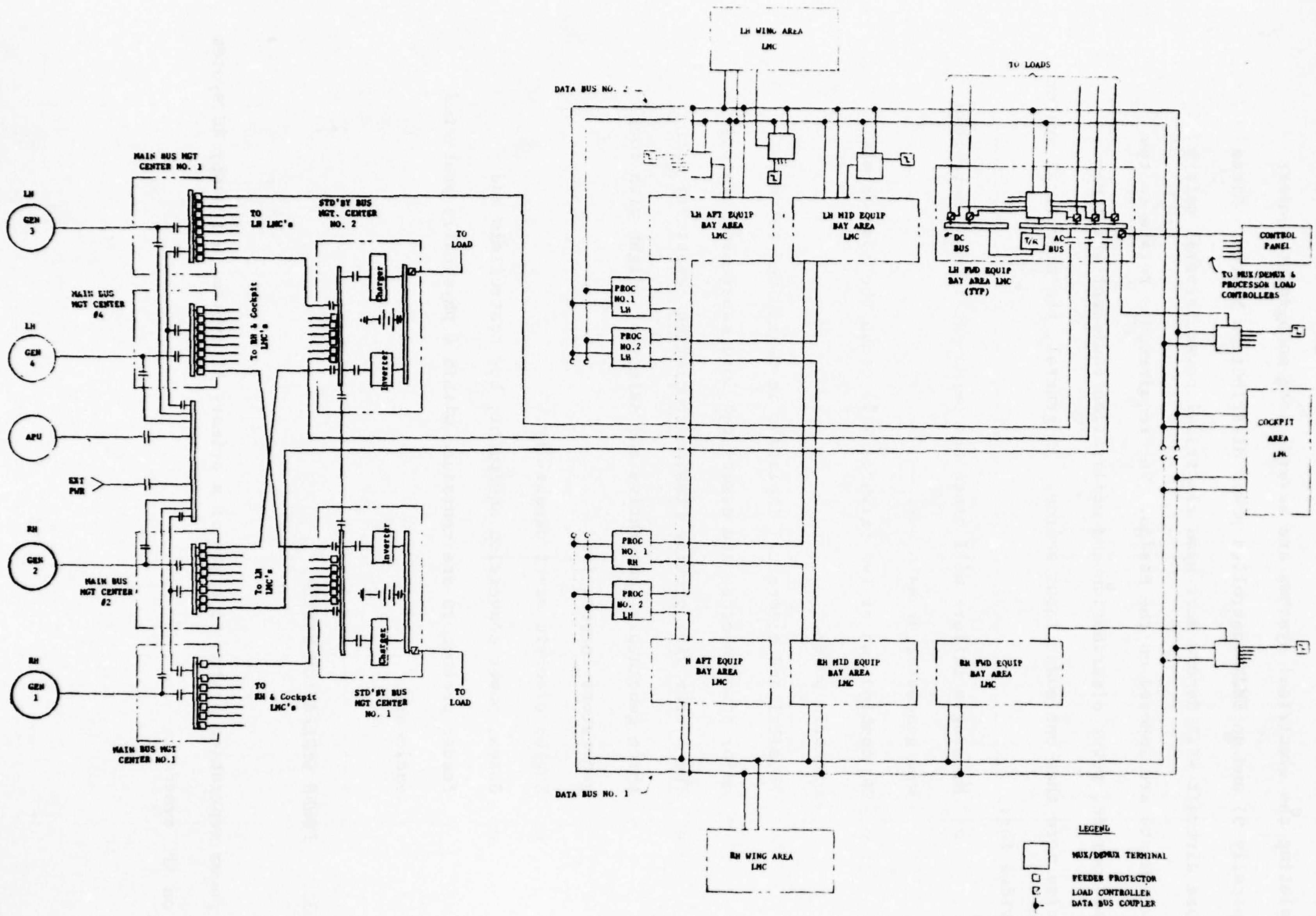

FIGURE 4. ADVANCED TECHNOLOGY E CTRIC CONTROL SYSTEM MULTI ENGINE AIRCRAFT

Completing the electrical systems are several load management centers (typically 9) and an EMUX controlled power distribution system. Since future aircraft will depend more upon electrical power, several safety requirements are imposed on the design. The requirements relate to the power sources, power distribution and utilization equipment which may require more than one power input source. In general, the electrical system provides for:

- No single failure will cause the permanent loss of more than one generator or main bus.
- No combination of two failures will cause the loss of all electric power.
- Physical and electrical isolation is maintained between major power sources and associated load management centers.
- The system will provide mission completion capability with three generators inoperative and safety of flight with four generators inoperative.
- Engine electric start capability.
- Buses, power conversion equipment, bus controllers and feeder protectors are contained within a physically protected enclosure.

4.2.1 POWER GENERATION SYSTEM

The power generation system consists of a primary AC system, a standby AC system and an APU system.

Primary AC System

The primary AC system consists of four advanced technology power generating channels (cycloconverter VSCF or IDG) which may be operated as either isolated, split parallel or full parallel. Synchronism between all four channels is automatically maintained in all operating modes whether isolated or parallel, consequently, no beat frequencies are present. System logic control is performed with microprocessors contained within each GCU. The GCU logic control provides the following capability:

- Each generator automatically comes on the line following engine start.
- Automatic load bus transfer upon a generator failure.
- Automatic lockout of faulted bus.
- Automatic lockout if output power is out of specified limits.
- Automatic protection for underspeed, under/over voltage, under/over frequency, out of synchronization, over current, DC current, waveform distortion and zero sequence voltage.

An interface to the EMUX system is provided for the purpose of obtaining data to improve operation (automatic load shedding) and maintenance, similar to that discussed for the single engine system. Additionally, integrated control techniques are employed through use of microprocessor implemented GCU and EMUX for providing generator and bus control functions.

4.2.1.1 MAIN DC POWER

Main 28 volt DC power is not provided on a system basis as projections indicate very little, if any, DC power will be required in the late 1980's. Specific loads requiring 28 volt DC power can be supplied from dedicated T/R units located within LMC's or from batteries as was discussed for the single engine aircraft.

4.2.1.2 STANDBY POWER SYSTEM

The standby power system is designed to supply MIL-STD-704 quality AC "gapless" power for the time duration it takes to bring the APU on line in the event of complete loss of primary AC power. The standby power system is a dual channel system with each channel consisting of a static inverter, state-of-the-art batteries, a battery charger, and power switching devices. Control for the solid state (or hybrid) standby bus controllers is obtained from the standby control unit (SCU). The SCU design and operation ensures that power interruptions to the standby AC buses do not exceed 20.0 milliseconds under worst case conditions. The SCU monitors the primary AC bus electrical characteristics for any loss of power. Should a power outage occur, the auxiliary AC bus controller is opened and the standby AC bus is switched to the operating inverter.

4.2.1.3 AUXILIARY POWER UNIT

The auxiliary power unit (APU) is capable of ground and flight operation, and supplies sufficient power to operate all the electrical loads including engine start. To aid in reducing the weapon system life cycle cost, a generator of the same type used for the main channels is used for the APU.

4.2.2 ENGINE ELECTRIC START

Engine start is provided with the use of either IDG or cycloconverter VSCF concepts. Electric power is supplied to an engine start bus from an external power source or an APU driven generator during the engine start function. The bus is used as a tie bus when power is supplied to the aircraft loads. Starter operation for the multi engine system is the same as discussed for the IDG and VSCF in paragraph 4.1.2 for the single engine aircraft.

4.2.3 POWER BUS MANAGEMENT

Power bus management consists of nine load management centers with each center supplied power with two main AC (3Ø) feeders and one auxiliary AC (3Ø) feeder. This provides triple redundancy for flight essential loads and dual redundancy for mission completion loads. Fuses, circuit breakers, RCCB's and electro-mechanical contactors are used to provide feeder fault protection and isolation. Each standby AC bus is supplied power from two main AC buses with cross-strapping between LH and RH sources. If the main buses experience a loss of power, the standby buses will supply power to flight essential loads for a specified duration via the battery powered inverters. Chargers are used to maintain charge on the batteries when power is supplied from the main or APU generators or from external power.

4.2.4 POWER DISTRIBUTION AND CONTROL

The power distribution system consists of signal sources, solid state power controllers and control with data multiplexing as was discussed for the

single engine aircraft. However, additional processors are provided to achieve high mission completion capability under the restrictions of data bus termination limitation. It is expected four processors in a split data bus arrangement will be required for a four channel power system.

Since four engine aircraft typically contains many more systems than a single engine aircraft, the power distribution system becomes more complex. This increased electrical system complexity is caused by the higher quantities of load controllers, signal sources, multiplex/demultiplex terminals and associated electrical harnessing.

Representative multiple (four) engine aircraft load controller and signal source quantities were derived from references 5 and 6 for P-3 and B-1 aircraft. In both instances, the number of load controllers required approached 1,000. Signal source transducer data was readily available for only the P-3 (reference 5). The approximate 470 mechanical or manual actuated signal sources for the P-3 is, however, representative of a large conventional military aircraft. It should be noted that the required signal source quantity is to a large extent, more dependent on the number of flight crew members than on engine count. This is especially true when considering the duplication of cockpit controls for the various flight crew stations. In addition, incorporation of multifunction cockpit control/display systems in the multiple engine aircraft will reduce the 470 discrete signal source count to approximately 320 transducers. The remaining 150 signals will be fed to EMUX over the controls/displays data bus. The estimate of remaining transducers (320) was derived by elimination of those switch functions defined in reference 5 which could easily be incorporated into multi-function cockpit control panels.

The number of EMUX multiplex/demultiplex terminals required is determined by the number of load controllers and signal sources transducers; the utilization efficiency of the terminals, channels, and the level of growth capacity desired. Based on studies reported in reference 2, the expected utilization efficiency of universal I/O EMUX terminals is 80 percent for 64 channel terminals. Assuming a 20 percent growth factor, the number of universal EMUX terminals required is:

$$\text{EMUX Terminals} = \frac{1.20\ (1000 + 320)}{.80 \times 63} = 31.43$$

Since a partial terminal is not available, a 32 terminal requirement is predicted. These 32 terminals along with four processors, a maintenance panel and a flight crew control/display panel form the basic EMUX control group. The processors will also monitor the AMUX buses to maintain a communication link between the electrical system and the various avionic subsystems. It is anticipated that capability will be provided for pilot and co-pilot to interrogate the EMUX processor through one of the multifunction control/display panels connected to the AMUX bus. It is recommended, however, that one EMUX dedicated (or primary function) control/display panel be installed at the flight engineer (or equivalent) station. This dedicated panel would permit continuous display of electrical system status. The dedicated EMUX panel could be implemented with a standard DAIS (Digital Avionic Information System) multifunction control/display panel with provisions for its' termination onto all four EMUX buses.

4.2.5 POWER QUALITY ASSESSMENT

Power quality is improved in the area of voltage and frequency transients since the power sources do not experience high load-on and load-off conditions. Faster

clearing of faults with solid state load controllers will also enable the transients to be minimized. The system contains "no break" power transfer to and from ground power. On start-up the transfer of power from ground power is smooth and automatic. When the first generator is brought up to speed, it synchronizes with ground power, takes over all the load and then disconnects the ground power. When the second generator is brought up to speed, it automatically synchronizes to the bus, parallels and takes over its share of the load. The two buses are then isolated so that each generator carries one half of the total load. The loads do not experience this power transition. Some of the specific power quality improvements expected are:

- Voltage regulation of better than ± 1 volt with up to 1/3 load unbalance.
- Voltage and frequency transients below the limits specified in MIL-STD-704.
- Frequency regulation with accuracies of a few parts per million.
- Load division within 5 percent of each generator.
- Reliability is improved due to the multiple redundant feeders supplying power from the sources to each load management center.

The significance of the improved power quality lies in the favorable impact on utilization equpment. Previous studies (reference 4) have addressed this improvement.

4.2.6 FLY-BY-WIRE SYSTEM

The baseline configuration for a multi-engine aircraft will provide electrical power with the integrity and reliability required by fly-by-wire systems. These requirements relate to the redundancy provided in the primary system (4 generators), cross ties between primary generator systems, three feeders to each LMC, four EMUX processors and a dual redundant data bus. The redundant "standby power center" minimizes the AC power interruption to 20 milliseconds maximum in the event the primary power source is momentarily lost. Power sensitive utilization equipment may require multiple power inputs to minimize the possibility of a power interruption in the event of a power source loss. The equipment would contain redundant power supplies with the outputs diode isolated.

SECTION V

TRADE STUDIES

Trade studies and relative assessments were made of various electric system capabilities that can provide significant benefits to aircraft electrical systems in the 1990 time period. Summaries are tabulated in Table 2. Generalized advantages and disadvantages may or may not apply to specific aircraft applications. Additional studies include power generating system trades, a reliability assessment on the power system, a reliability assessment on EMUX processor redundancy, EMUX integrated power control concepts and a weight evaluation of engine electric self start capability. These studies were conducted to support selection of preliminary designs for a single engine and a multi-engine aircraft.

5.1 POWER GENERATING SYSTEM STUDY

A trade study was conducted on the contending advanced technology generating concepts. The concepts include the VSCF (cycloconverter and DC link), CSD and IDG systems. Trade parameter data was received from generating system manufacturers in the form of a response to Vought's "Request for Data Questionnaire". Table 3 lists the questions submitted by Vought and the manufacturers response to each question. It should be noted that the questions and responses are generalized. Actual numerical values will vary in a specific aircraft application due to design constraints imposed by the prime aircraft developer. For this reason, direct comparison of the numerical data must be made cautiously.

The conclusion is that generating system requirements for the 1990 time period aircraft can be met with either the IDG or the VSCF (cycloconverter) concept.

TABLE 2

ELECTRIC SYSTEM CAPABILITIES

FUNCTION	ADVANTAGES	DISADVANTAGES
Electric Engine Start (Starter/ Generator)	1. Eliminates the need for a separate starter on each engine. 2. Engine frontal area is reduced since conventional starter is eliminated. 3. Some of starter weight can be in avionics bay (converter) rather than on the engine. (Not applicable to integrated VSCF concept.) 4. Does not require quality electric power, i.e., supply voltage can droop at the beginning of the start cycle. 5. Savings in weapon system realiability and cost are anticipated over the separate starter concept.	1. Does not in itself provide "self start" capability. 2. 400 HZ power must be available. 3. Use of battery/inverter for "self start" capability is not practical due to large quantity of power required. 4. Power supply output may be distorted by the starter during starting mode and may not be suitable for use by aircraft avionic loads during the start cycle. (Not applicable to IDG systems.) 5. Increases generator system weight approximately 12% per channel due to additional controls required. 6. Will increase electric system failure rate due to added control complexity and added stress on the system. 7. Redesign of existing external power cart is required. 8. Generating system rating is usually dictated by start requirements rather than electric load requirements. (Approx. 90 KW or greater is required to start most engines)

ELECTRIC SYSTEM CAPABILITIES (Continued)

FUNCTION	ADVANTAGES	DISADVANTAGES
Electric Power Under Engine Windmilling Conditions	1. May eliminate the need for a separate emergency power source.	1. Adds approx. 8% to generating system weight due to the added controls required.
	2. Approximately 12% of generator rating can be converted to 28 VDC at 25% of max. rated engine speed.	2. Only "variable frequency" power can be generated, i.e., AC power not suitable for most aircraft loads (an inverter is required)
		3. Does not take the place of an APU (Does not provide ground power)
		4. Must be disabled for certain failure modes.
Paralleling Capability	1. Provides a stiffer power source, consequently lower transients are generated during heavy load switching.	1. Adds approx. 3% to generating system weight per channel due to additional controls required.
	2. Can provide an uninterrupted power bus during bus switching or in the event of a generator channel failure. Power is interrupted only for certain channel failure modes.	2. "Stiffer" power source not likely required with load management, but may be required for a specific load.
		3. Control of system stability is more complex.
		4. All parallel busses interrupted or disturbed for fault clearing or certain channel failure modes.
Automatic Load Management	1. Prevents possible loss of generating system due to a malfunction or component degradation	1. May not be needed in a multi-channel system due to the redundancy provided.
	2. Can be provided by EMUX with essentially no weight or cost penalty.	2. Control of system stability is more complex.
		3. Decrease in system reliability due to added complexity.

ELECTRIC SYSTEM CAPABILITIES (Continued)

FUNCTION	ADVANTAGES	DISADVANTAGES
Automatic Load Management (Cont'd.)	3. Appropriate for use on an APU to provide optimum use of available power since APU rating varies with altitude, cooling, etc. 4. Predominate advantage in multi-channel system results from improvement in mission completion capability during multiplex failures. 5. Reduces level of step power changes in loads, resulting in lower voltage transients. 6. Can possibly reduce generator system rating, thus reduce weight. 7. Can be used to limit load on APU during ground operations and to automatically "safe" specific loads during ground checkout.	
Perform GCU Function in EMUX	1. A weight savings is realized over systems having GCU as a separate assembly. 2. Simplifies interface wiring in multi-channel systems, i.e., paralleling, synchronizing, load division requirements, etc. 3. Provides all control data at central location.	1. GCU function in VSCF systems is performed within converter. Minimal weight savings is realized with removal of GCU functions. 2. EMUX "response time" is not compatible with realtime generator performance requirements. 3. Possible loss of effective quad redundancy in a four channel system. 4. Complicates and increases control power requirements for system start-up. (Power for GCU function must be derived from a separate source) 5. Correlation of GCU requirements between two vendors is required (Generator vendor and EMUX vendor)

ELECTRIC SYSTEM CAPABILITIES (Continued)

FUNCTION	ADVANTAGES	DISADVANTAGES
28 VDC Power Supplied Directly From Variable Speed Generator	1. Decrease system weight since T/R magnetic components are small due to high frequency. 2. Loss of regulated AC power does not result in loss of DC power for most channel failure modes (AC converter is in parallel with DC inverter)	1. Separate regulator is required for DC power. 2. Additional contactor is required (between T/R unit and load bus) 3. Certain channel failure modes result in loss of both the AC channel and the T/R associated with it. 4. Provisions must be made for DC when operating with ground power.
Permanent Magnet Rotor Generator	1. Eliminates the need for a field winding and associated cooling requirements. 2. Eliminates the need for rectifiers and associated cooling requirements. 3. Eliminates the need for an auxiliary power supply. 4. Anticipated increase in reliability due to the elimination of windings, rectifiers, and cooling provisions. 5. Total losses are lower than in wound rotor generator. 6. Generator efficiency is higher due to absence of exciter losses and lower windage losses resulting in higher overall efficiency. 7. Simplifies generator/starter design requirements.	1. Generator creates higher voltage variations since field excitation cannot be controlled. 2. Voltage regulation control in converter is more complex. 3. Converter efficiency is lower due to higher losses in despike networks 4. SCR voltage ratings must be higher, consequently more costly. 5. A disconnect (electrical or mechanical) is required to protect from certain failure modes since the generator cannot be de-energized by removing excitation.

ELECTRIC SYSTEM CAPABILITIES (Continued)

FUNCTION	ADVANTAGES	DISADVANTAGES
EMUX Power Control	1. Allows weight reduction in complex electrical systems. 2. Improves maintainability. 3. Decreases life cycle cost. 4. Improves power quality (loads are programmed ON and OFF)	1. Only practical for moderate to very complex (numerous loads) electrical systems. No size and weight advantage for simple systems.

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TABLE 3

DATA SURVEY — ADVANCED TECHNOLOGY ELECTRIC POWER SYSTEM (SHEET 1 OF 4)

VOUGHT QUESTION	WESTINGHOUSE	GENERAL ELECTRIC	SUNDSTRAND
1. What power generation concept do you envision will best meet the needs of aircraft electric power systems in the 1980 1990 time period?	VSCF (cycloconverter) 115/200 V, 400Hz	VSCF (cycloconverter) 115/200 V, 400 Hz	Integrated Drive Generator 115/200 V, 400 Hz
2. What is weight of the above system (per channel) from engine mount to point of regulation? (point of regulation is 10 ft from power out terminals)	TODAY Rating (kVA) 40.0 60.0 90.0 150.0 Drive — — — — Generator 45.0 52.0 67.0 95.0 Converter 40.0 55.0 65.0 80.0 GCU 5.0 5.0 5.0 5.0 CT's 0.8 1.0 1.25 1.6 Feeders 8.3 12.8 20.0 33.0 QAD 3.0 4.0 4.0 4.0 Oil/Cool. Prov. 10.0 11.0 12.7 16.0 Other — — — — Total (lb) 112.1 140.8 174.9 134.6 PROJECTED (1990) Rating (kVA) 40.0 60.0 90.0 150.0 Drive — — — — Generator 40.0 47.0 60.0 85.0 Converter 35.0 50.0 60.0 75.0 GCU 5.0 5.0 5.0 5.0 CT's 0.8 1.0 1.25 1.6 Feeders 8.3 12.0 20.0 33.0 QAD 2.8 3.8 3.8 3.8 Oil/Cool. Prov. 9.5 10.5 12.0 15.0 Other — — — — Total (lb) 101.4 130.1 162.0 218.4 Cycloconverter cooling not included Feeders Established by Vought Gen Cooling Includes Cooler and Coolant Pump	TODAY Rating (KVA) 40.0 60.0 90.0 150.0 Drive Generator 49.0 65.0 85.0 110.0 Converter 45.0 57.0 75.0 96.0 GCU CT's 1.0 1.0 1.0 1.0 Feeders 8.3 12.8 10.0 33.0 QAD 1.0 1.0 1.0 1.0 Oil/Cool. Prov. Other Total (lb) 104.3 136.8 182.0 241.0 PROJECTED (1990) Rating (KVA) 40.0 60.0 90.0 150.0 Drive Generator 42.0 59.0 78.0 103.0 Converter 40.0 50.0 65.0 85.0 GCU — — — CT's 1.0 1.0 1.0 1.0 Feeders 8.3 12.8 20.0 33.0 QAD 1.0 1.0 1.0 1.0 Oil/Cool. Prov. Other Total (lb) 92.3 123.8 165.0 223.0 Feeders Established by Vought Converter Cooling Not Included GCU Included in Converter Gen Cooling Shared With Gearbox	TODAY Rating (kVA) 30 40 60 75 75 90 120 150 Drive 34.0 40.0 57.0 105.0 Generator 22.5 33.5 53.0 74.0 Generator 22.5 33.5 53.0 74.0 Converter GCU 4.5 4.5 4.5 4.5 GCU 0.82 0.82 1.3 1.3 Feeders 8.3 12.8 20.0 33 QAD 4.1 4.1 4.1 4.1 Oil Cool Prov 0.8 1.0 1.2 1.65 Other Total (lb) 75.0 96.7 141.1 223.5 PROJECTED (1990) Rating (kVA) 30 40 60 75 75 90 120 150 Drive 32.0 36.0 51.0 94.0 Generator 17.5 25.0 38.6 54.0 Converter GCU 3.0 3.0 3.0 3.0 CT's 0.5 0.5 0.75 0.75 Feeders 8.3 12.8 20.0 33.0 QAD 2.0 2.0 2.0 2.0 Oil/Cool Prov 0.8 1.0 1.2 1.65 Other Total (lb) 64.1 80.3 116.6 188.4 Feeders Established by Vought
3. Does system have capability of supply electric power under engine windmilling conditions? If "yes", define output power vs engine rpm 10% max. rated rpm 20% max. rated rpm 30% max. rated rpm 50% max. rated rpm	Yes — output power is 115/200 V Today / Projected 0% / 0% 0% / 0% 0% / 0% 100% / 100%	Yes — output power is 30 V dc Today / Projected 10% / 20% 20% / 25% 25% / 25% 25% / 25%	Yes — output is assumed to be converted to 28 V dc Today / Projected 0% / 0% 42% / 42% 65% / 65% 100% / 100%
4. Does system provide engine start capability? If "yes", (a) How much weight (in %) is added to the basic system? (b) What are torque/rpm characteristics?	No (being studied)	Yes (a) Add 10% basic system weight (b) Similar to dc machine	Yes (a) 40 kva 6.5 lb 60 kva 7.5 lb 90 kva 8.0 lb 150 kva 15 lb (b) Per torque/speed curve
5. What magnetic material is used for the generator?	HYPERCO 27 — Today HYPERCO 50 — Projected (1990)	Permendur — Today Projected	Cobalt/vanadium/iron — Today Cobalt/vanadium/iron — Projected
6. What is recommended speed? (assume 2 to 1 range)	26,000 RPM — Today 26,000 RPM — Projected (1990)	22,000 27,000 — Today 24,000 36,000 — Projected	18,000 rpm — Today 24,000 rpm — Projected

TABLE 3

DATA SURVEY ADVANCED TECHNOLOGY ELECTRIC POWER SYSTEM (SHEET 2 OF 4)

VOUGHT QUESTION	WESTINGHOUSE	GENERAL ELECTRIC	SUNDSTRAND
7 What is system efficiency to point of regulation? At min speed rated speed max speed	Today / Projected KVA 400 600 900 1500 / 400 600 900 1500 76 8 79 7 82 6 84 5 / 77 8 80 6 83 5 85 4 70 1 73 9 77 8 81 6 / 71 0 74 9 78 7 82 6 63 4 68 2 73 0 77 8 / 65 3 69 1 73 9 78 7 Does not include feeder losses	Today / Projected KVA 400 600 900 1500 / 400 600 900 1500 80 0 82 0 84 0 85 0 / 82 0 84 0 85 0 88 0 79 0 81 0 83 0 84 0 / 80 0 82 0 84 0 86 0 78 0 80 0 82 0 83 0 / 79 0 81 0 83 0 84 0 Does not include feeder losses	Today / Projected KVA 30 40 60 75 75 90 120 150 / 30 40 60 75 75 90 120 150 80 0 80 0 80 0 80 0 / 80 0 80 0 80 0 80 0 81 0 81 0 81 0 81 0 / 81 0 81 0 81 0 81 0 77 8 77 8 77 8 77 8 / 77 8 77 8 77 8 77 8 Does not include feeder losses
8 Does system have parallel operation capability? If "yes", how much weight (in percentage) is added to the basic system?	Yes Approx 3 lb	Yes Approx 3 lb	Yes Appox 1 0 lb per channel
9 What overload capability does the system provide for 5 minutes, 2 minutes, 5 seconds?	1 5 PU for 5 minutes 2 0 PU for 5 seconds or whatever is required	1 5 PU for 5 minutes 2 0 PU for 5 seconds	1 5 PU for 5 minutes 1 7 PU for 2 minutes 1 75 PU for 5 seconds Can provide "whatever is req'd" at some increase in weight
10 Is power quality better than Mil Std 704? If "yes", in what areas and how much? (a) Voltage Regulation (b) Voltage Transients (c) Distortion (d) Frequency Regulation (e) Frequency Drift (f) Frequency Transients (g) Overvoltage Transients (h) Undervoltage Transients (i) Phase Voltage Balance	Yes Today / Projected ±0 8% / No Change Recovery in 10ms / Recovery in 10ms Less than 5% / Less than 4% ±0 1% / No Change ±0 1% / No Change None / None 160 VRMS / No Change 90 VRMS / No Change ±1 5V (120±2 5°) / No Change	Yes Today / Projected ±1 0 V / ±1 0 V See Mil E 23001B 4% / 4% <1 0 Hz / <1 0 Hz <1 0 Hz / <1 0 Hz None / None See Mil E 23001B See Mil E 23001B Individual Phase Voltage Reg	Yes Today / Projected ±1 0 V / Meet or exceed req'mnt of Mil G 21480 & Mil Std 704 Recovery in 15 ms 2% ±4 0 None ±20 Hz 135 V 15 90 vrms 3 5% @ 2/3,0,0 PU
11 Is microprocessor used in the GCU to improve capability? If "yes", what improvements are expected?	Today — No Projected - Yes Microprocessor will improve self test and provide flexibility Will not improve power quality	Today - No Projected - Yes Microprocessor will improve maintainability	Today - Yes Projected - Yes Microprocessor improves self test capability and provides better steady state regulation
12 Is DC component of voltage present on the 400 HZ bus? If "yes", how much?	Yes Less than 50mv	Yes Less than 50mv	No
13 To what extent can built in test and fault isolation be provided? What parameters should be monitored?	Any reasonable level of built in test can be provided Can provide fault isolation to LRU	Fault isolation to LRU	Annunciates continuous and intermittent faults in IDG, GCU, BCU, feeders, buses and control wiring independent of circuit reliability
14 What is minimum service life between overhaul? For drive generator converter GCU	Today / Projected — / — On Condition / On Condition 2,000 hours / On Condition On Condition / On Condition	Today / Projected — / — On Condition / On Condition On Condition / On Condition On Condition / On Condition	Today / Projected On Condition / On Condition On Condition / On Condition — / — On Condition / On Condition
15 What type of cooling system is used for the drive? Generator? Converter?	Drive / Gen / Conv Type of Coolant — / Oil / Oil Coolant Temp — / 250°F / 176°F Coolant Flow (gpm) — / 2 7 / 2 10 Coolant Pressure (psi) — / 50 250 / 200 Coolant Capacity — / 0 75 pt / 1 3 qt *Depends upon KVA rating	Drive / Gen / Conv Type of Coolant — / Oil / Oil/Air Coolant Temp — / 125°C / 71°C Coolant Flow — / 50 pm / 50 pm Coolant Pressure — / 100 psi / 100 psi Coolant Capacity — / 2 qt / 2 qt	Drive / Gen / Conv Type of Coolant Oil / Oil / — Coolant Temp (°F) 65÷300 / 65÷300 / — Coolant Flow (gpm) 8 / 2 7 / — Coolant Pressure (psi) 60 / 60 / — Coolant Capacity (qts) 3 / ½ / —

TABLE 3

DATA SURVEY   ADVANCED TECHNOLOGY ELECTRIC POWER SYSTEM (SHEET 3 OF 4)

VOUGHT QUESTION	WESTINGHOUSE	GENERAL ELECTRIC	SUNDSTRAND
16 **Would the same coolant system be used independent of generator system ratings?**	Yes	Yes 12 GPM depending on rating	Yes
17 **What type of technology is recommended for the bus controller (line contactor) function? Solid state, electromechanical or hybrid?**	Electromechanical	Electromechanical	Electromechanical
18 **What control power characteristics (voltage & current) are recommended for the bus controller?**	8 to 20 vdc 4 to 16 watts	28 vdc 2.0 amps	15 to 56 volts 2 to 15 amps
19 **Will system meet requirements of Mil Std 461?**	Yes	Yes	Today Yes Projected Yes
20 **What is MTBF and MTTR for?**	**MTBF (hours)** Today / Projected	**MTBF (Hours)** Today / Projected	**MTBF (hours)** Today / Projected
Drive	—		5,200 / 20,000
Generator	12,000 / 24,000	15,000 / 30,000	13,000 / 20,000
Converter	5,200 / 10,000	10,000 / 20,000	
GCU	30,000* / 33,000*		4,000 / 10,000
	MTTR (hour)** Today / Projected	**MTTR (hours)** Today / Projected	**MTTR (hours)**** Today / Projected
Drive			20 / 10
Generator	8 / 8	7 / 5	6 / NA
Converter	3 / 2.7	3 / 2	
Gcu	2 / 1.8		4 / 3
	**Performed at depot *Improved LSI parts		**Performed at depot, includes drive and generator as one package
21 Does system have capability of operating isolated and synchronized? If "yes", (a) What is weight penalty for this feature? (b) What is the practical maximum no. of generating channels which can be synchronized?	Yes (a) None (b) Eight	Yes (a) 1% of basic system weight (b) Unlimited	Yes (a) One pound per channel (b) No limit
22 What is the maximum fault current (% rated) and time duration prior to gen trip?	2 PU current for 10 seconds (can be designed to meet any requirement)	3 PU current for 5 seconds	3 to 4 PU current for 10 seconds
23 Is it feasible to reduce fault current levels prior to opening the bus controller? (To prevent the bus controller from having to interrupt high fault currents). If "yes", will this control be sufficiently reliable to permit the use of a smaller bus controller?	Yes (a) Yes, this type of control is presently employed in cycloconverters to limit current	Yes (a) Yes	Yes
24 What is the projected "optimum" nominal voltage level and frequency of the generating system for the 1980-1990 time period? (i.e. 115/200 V, 400 Hz, 800 Hz, 230/400 V, 400 Hz, 270 V dc, other)	115/200 V, 400 Hz 800 Hz is too high for converter 230/400 V is too high for SCR to be used in converter 270 V dc not practical for near future because of development work needed	115/200 v ac, 400 Hz	115/200 V ac 400 Hz 230/400 V ac 400 Hz

TABLE 3

DATA SURVEY — ADVANCED TECHNOLOGY ELECTRIC POWER SYSTEM (SHEET 4 OF 4)

VOUGHT QUESTION	WESTINGHOUSE	GENERAL ELECTRIC	SUNDSTRAND
25 In a four generator system, should each generator be operated (partially loaded) continuously or should some generators be held in standby (no load but rotating)? What is relationbship between MTBF and loading as a percent of rated load? What is relationship between efficiency and loading as a percent of rated load	Operate each generator continuously At full load, MTBF is approx 75% of MTBF at half load Rate of no load losses to full load losses increases with increase in speed	Very feasible Very little effect with VSCF *Highest efficiency at rated load* Half load is three points less, at quarter load, 14 points less	Operate all channels under load. As it will provide balanced life, reliability and smaller voltage transients At cruise speed the efficiency at ½ rated load is approx 11% less than at rated load. The efficiency at ¼ load is *approx* 5½% *less than at* ½ *load*
26 Do generator system failure modes frequently occur which *permit the generator to deliver* power, but at reduced level? If "yes", what generator parameters could be monitored to detect these conditions?	Generally no, however, open field diode limits overload rating and shorted field diode limits loads to 25% rated load Same effect for shorten and open leads in exciter armature	No	No
27 Should multichannel system load division be controlled by the associated GCU's or by a separate load division controller?	Within each cycloconverter	Associated GCU's	Associated GCU's
28 What percent of generator failure rate can be attributed to rotation alone? That is, what is the difference in MTBF between a standby generator operating in a no-load but rotating condition vs operating in a nonrotating *condition? If the MTBF difference is significant,* is a mechanical clutch which would permit connection of a generator to a rotating engine pad output feasible? How much weight and MTBF penalty would such a clutch impose on gen ?	60% *of failure rate is attributed to gen rotation* Mechanical clutch is feasible. It would add 8% of gen weight to system MTBF on clutch is not available	20% for wound rotor and 5% for solid rotor Yes in a few years Adds 3 to 5 lbs of weight Approx 15% *reduction in MTBF* MTBF is dependent upon the no. of actuators	55% of gen failures is attributed to rotation Clutch is feasible but is very unreliable and heavy
29 What is recommended design implementation for providing no-gap or minimal gap power under conditions of "primary system failure" and stand by or emerg power systems not *operating? What is recommended "maximum"* allowable gap based on GCU & hardware response?	Operate systems in parallel However a single fault could impact all systems Sense failure, open controller to isolate fault, sense bus and close bus controller	Supply dc power to critical loads	Sense fault, transfer to standby source, startup alternate source, transfer to alternate source
30 What advantage/disadvantages are projected for an integrated engine gen system? Is a permanent magnet gen a prerequisite for this type design? Will this dictate a preferred type of power, i.e. dc, high freq ac, etc.	Feasibility not known at this time If feasible, a weight saving should be realized Time & cost of servicing would increase Permanent magnet gen not prerequisite	Feasible with permanent magnet gen Not dictated by type of power	Feasible with either wound rotor or permanent magnet Permanent magnet preferred because of high reliability, however quick automatic disconnect req'd to limit fault current
31 Based on the "advanced power generation-control" system recommended, *it is believed to* be feasible to have the EMUX system perform the GCU functions? What is the max. "transport delay" allowable in the EMUX for providing an "error correction" signal in *response to a sensed "error" signal for voltage* and frequency regulation, fault clearing, etc ?	Using EMUX to perform GCU function is not practical *EMUX delay of less than 5 millisedonds is required*	Not feasible	Feasible if EMUX transport delay is less than 100 micro seconds

Specific aircraft requirements will generally establish the concept to be used. An advanced IDG design offers some advantages over the cycloconverter VSCF system in the areas of size and weight. A weight comparison of typical systems is shown in Figure 5. It is noted that the data submitted for the IDG are for 1.75 P.U. overload rated systems. A 2.0 PU rating will result in some increase in weight, although it still will be below the VSCF weight. A description of an advnaced technology 90 KVA system is given in Appendix A. The IDG system also provides a high degree of confidence in terms of low technical risk since it is a design of mature technology. The cycloconverter VSCF system offers some improvements in power quality (precise frequency, no frequency transients, short duration voltage transients) and a potential for higher reliability, weight reduction and lower life cycle costs. These attributes are primarily in the converter area where frequency conversion and voltage regulation are performed with solid state components. Manufacturer supplied data (see Figure 5) also indicates a smaller weight increase per KVA as system ratings increase. Lowering power dissipation and improving semi-conductor cooling techniques are the key to optimizing the VSCF system since semiconductor reliability is highly influenced by temperature.

The CSD (drive separate from generator) is not considered a candidate system for new aircraft designs even though these systems are operating on present day aircraft. The CSD system does not provide the desired low weight of either the IDG or VSCF systems. Also, the DC link VSCF concept has not progressed enough to be considered a viable candidate system for the 1990 time period. The High Voltage DC (HVDC) generating system was not

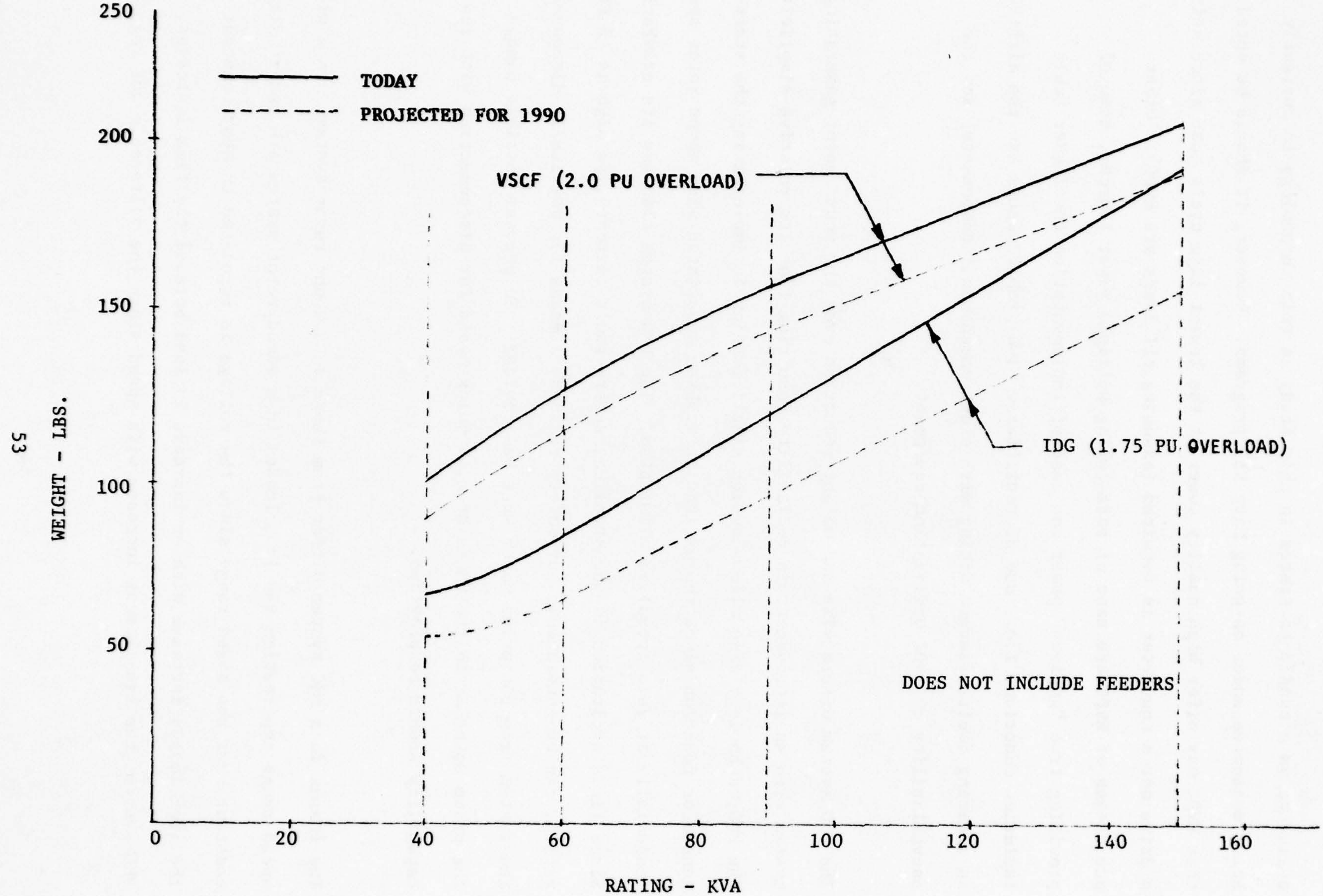

FIGURE 5 TYPICAL GENERATING SYSTEM WEIGHT COMPARISON

evaluated as a candidate system in this study as this technology is currently being evaluated under on-going Navy R&D programs. However, it should be noted that HVDC may offer high quality power at the lowest life cycle cost since neither a drive nor a converter is required (assuming all loads are HVDC). Other advantages of HVDC are ease of paralleling multiple power sources, ease of providing true "gapless" power and ease of implementation the feeder fault isolation function, i.e., use of rectifiers. Two problem areas are the difficulty in clearing faults (severe arcing with electromechanical components) and the unavailability of HVDC utilization equipment.

The PMG system offers size and weight reduction over the wound rotor generating system with an improvement in reliability resulting from its relative simplicity. The PMG technology also allows design simplification in implementing the starter/generator function since the excitation problem associated with wound rotor systems (especially at zero speed) is eliminated. The PMG system also has its drawbacks. Since field excitation is uncontrolled, severe fault currents, as high as 15 PU rating, can be generated. For safety reasons, a means for physically disconnecting the system from the prime mover must be provided. To prevent possible damage to the prime mover or to the PMG system, a quick means for disconnecting with reset capability should be provided.

The losses in a PMG system differ from those in a wound rotor system. In a wide speed range application the iron losses in a wound rotor system are essentially constant over the speed range since the voltage is regulated to stay constant. Also, the iron losses increase with an increase in load because the flux increases. In the PMG system, the iron losses increase with speed since the voltage is not regulated,

and decreases with an increase in load since the flux level is reduced by armature reaction. The total losses (windage, excitation, iron) are less for a PMG system than for a wound rotor system.

Low voltage DC power (28 VDC) was the primary electrical power source on early aircraft. As utilization equipment power demand increased, the demand for 28 VDC power relative to AC power has diminished to a level that an all AC system can be projected for the 1990's. Obviously, some 28 VDC power will still be required in the 1990's but this will be for "special" load conditions which can be supplied more efficiently than on a system basis. Typically, this can be accomplished as follows:

> In the first approach, a conventionally hardwired power connection can be provided from a battery to the appropriate load such as a communication set "black box". The power wire would be protected by a "normally closed" solid state power controller. This permits delivery of power independent of EMUX while allowing control of the SSPC after EMUX is powered up. The battery used as the power source is the same battery used to start the APU and to provide a short term "uninterrupted" power bus.
>
> The second approach is to use self contained batteries for a backup power source. In this concept, a small rechargeable battery is installed in (or adjacent to) the associated black box. When available, bus power is supplied to the equipment from the EMUX controlled source. This self-contained battery approach is acceptable

as long as the number of loads requiring back-up power is small and other operational requirements do not dictate installation of a large battery.

A third concept is to install transformer rectifier units within load management centers to power "localized" DC loads. Here again, it is assumed the DC load demand is low.

An aircraft with either fly-by-wire or engine self-starting (or both) will virtually ensure the installation of a reliable, instantaneously "energized" backup power source. These characteristics are difficult to achieve unless the source is DC. However, it is advantageous from an overall system integration viewpoint to convert this power to AC.

By standardizing <u>all</u> bus management and power distribution hardware on either AC or DC, significant savings can be achieved in electrical system life cycle costs. These cost savings result from:

- Lower system weight by eliminating secondary power converters, feeders, buses and switching hardware.
- Increased reliability and reduced maintenance actions, through elimination of the above equpments.
- Lower logistic cost by eliminating the secondary power hardware and reducing the number of different load controller types.
- Permits sufficient standardization of load controllers to simplify the Integrated Load Management Center concept.

These benefits can be achieved by standardizing on either 115 or 230 volts AC or on a comparable DC voltage level.

The potential advantages to standardization on DC power lie predominately in lower avionic power supply weights and simplified power source isolation and paralleling. Acceptance of high voltage DC at this time, however, is limited by industry confidence in low technical risks. Since minimal risks are foreseen from selection of an AC standard, an AC system is best for the preliminary design.

The major power generation subsystem question which still remains is the impact of fly-by-wire reliability and vulnerability requirements. These requirements can only be tailored once the aircraft system performance characteristics have been defined. At this point, the best study approach is to define the level of reliability available with the selected power system and provide options for improvement if the levels are considered insufficient.

5.2 RELIABILITY ASSESSMENT OF POWER SYSTEM

A study was conducted to determine the significant parameters which impact the electrical system reliability. This was done by allocating the system failure rate among three major subsystems; power generation, bus configurations and power distribution. It should be noted that the requirements given in sections 3.1 and 3.2 for mission completion and safe return of aircraft were goals to be achieved. Furthermore, the requirements should be considered as typical since specific aircraft missions will dictate specific reliability requirements.

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The reliability requirements (see 3.1 and 3.2) for delivering power to a bus and to all utilization equipment on a single and a multiple engine aircraft summarized as follows:

		Probability of Success	
		Single Engine	Multi-Engine
Aircraft Safe Return	- Bus	.9998	.99995
	- All Equip	.998	.991
Mission Completion	- Bus	.995	.9998
	- All Equip	.990	.980

5.2.1 POWER TO BUS

A reliability assessment was made on several power generating system configurations to establish the minimum number of generating channels required on a multi-engine (4) aircraft to meet the required reliability goals. Since the number of channels is primarily influenced by mission completion requirements, the assessment provides insight into an optimum arrangement. Figure 6A through 6H depict reliability block diagrams for the significant (failure prone) hardware of eight configurations for a four engine aircraft. The block diagrams illustrate the primary AC and, when applicable, secondary DC portions of the main power system. The top reliability block of each diagram represents a series connection of the turbofan engine (MTBF $\approx$ 600 Hrs) and the AC generator (MTBF $\approx$ 1000 Hrs)

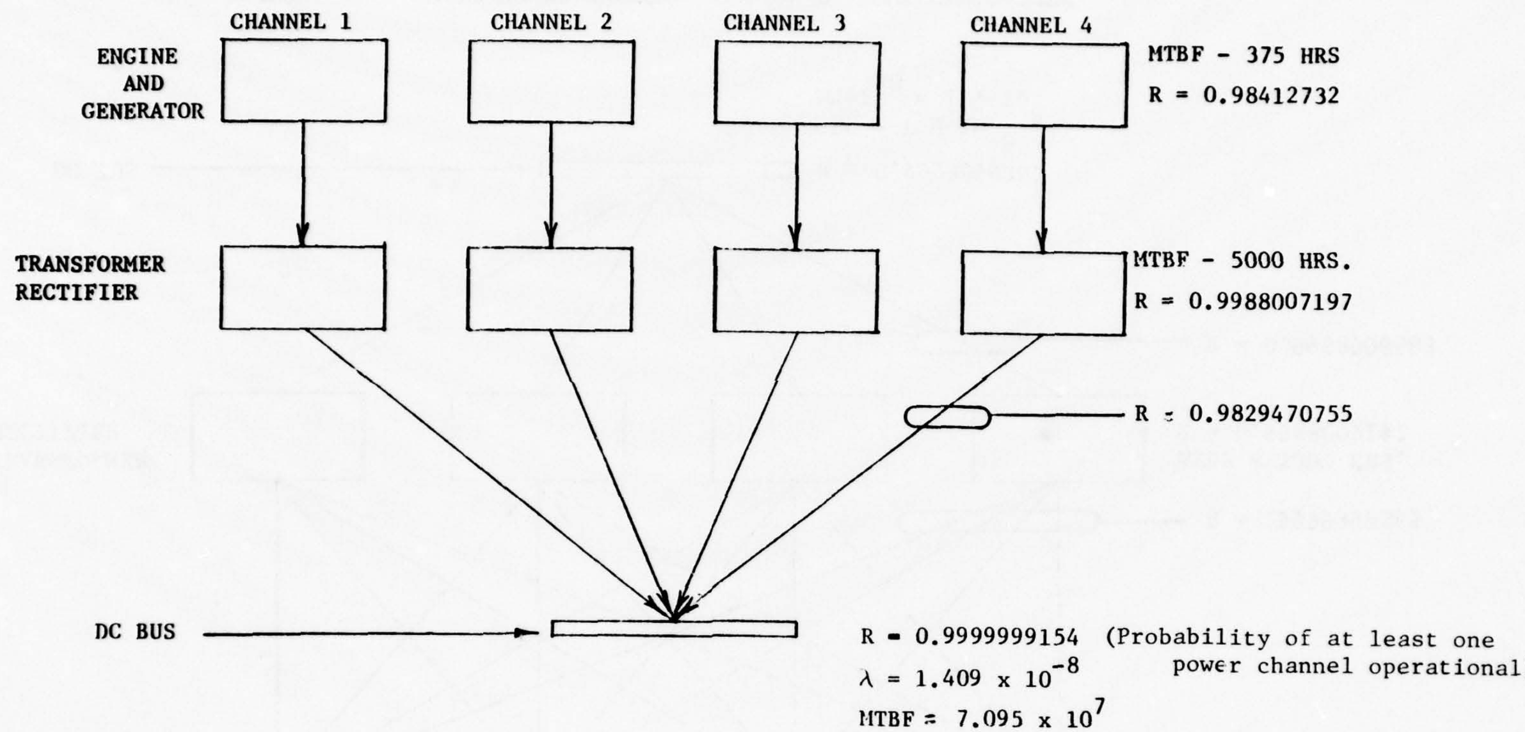


FIGURE 6A - MAIN POWER SYSTEM - OPTION 1 - FOUR GENERATORS, FOUR T/R UNITS, NO AC CROSS-STRAPPING

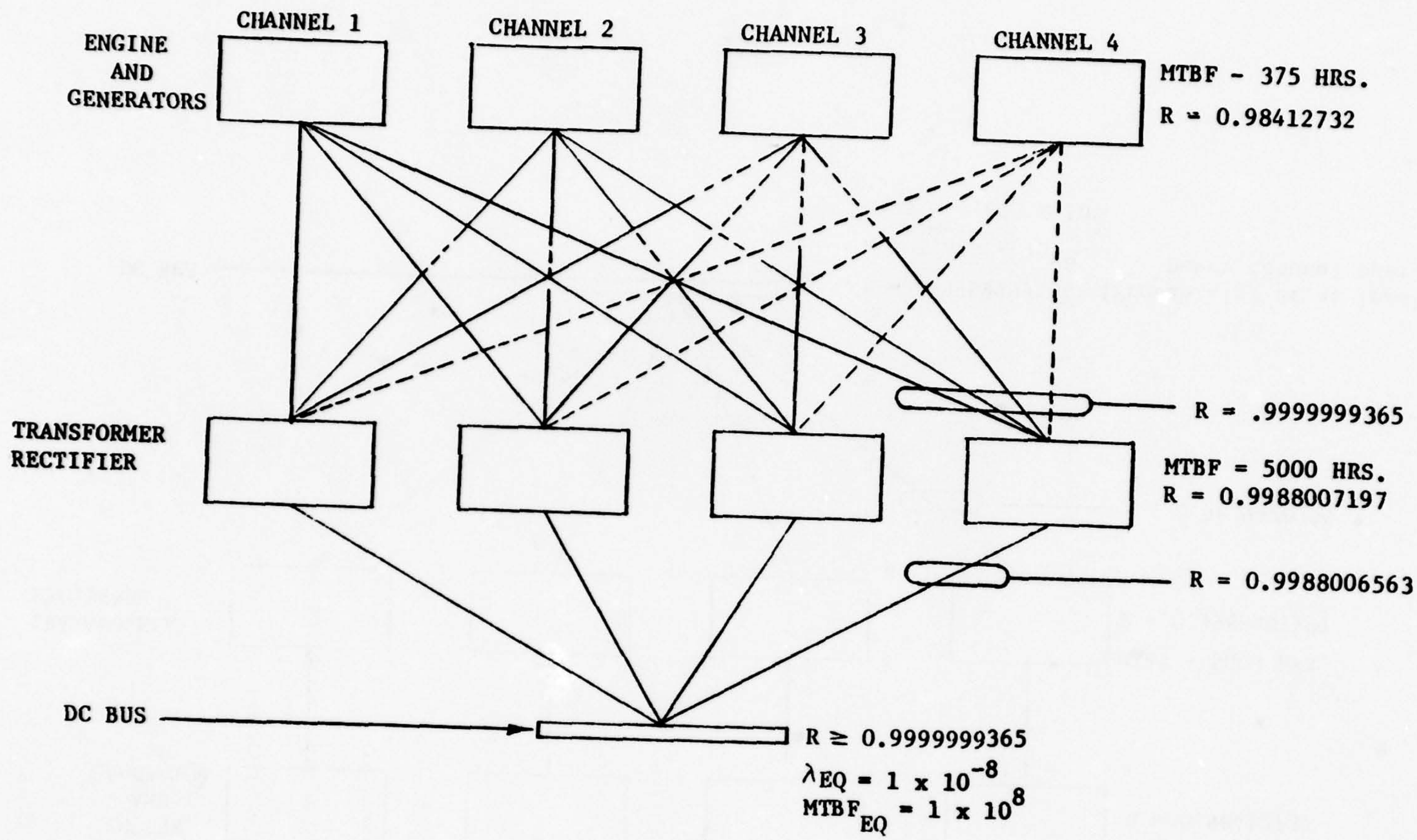


FIGURE 6B MAIN POWER SYSTEM - OPTION 2, FOUR GENERATORS, FOUR T/R UNITS, AC CROSS-STRAPPING

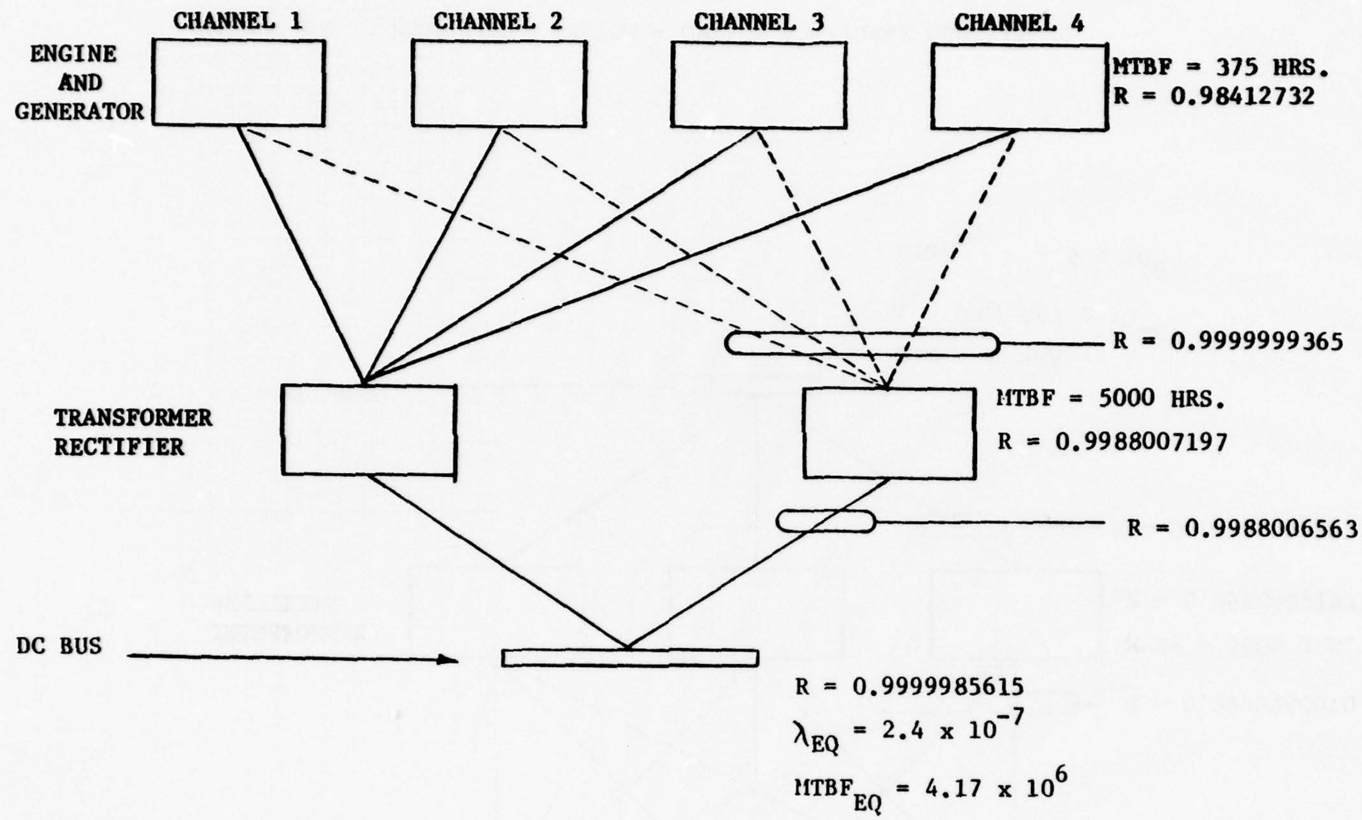


FIGURE 6C MAIN POWER SYSTEM - OPTION 3 ~ FOUR GENERATORS, TWO T/R UNITS, AC CROSS-STRAPPING

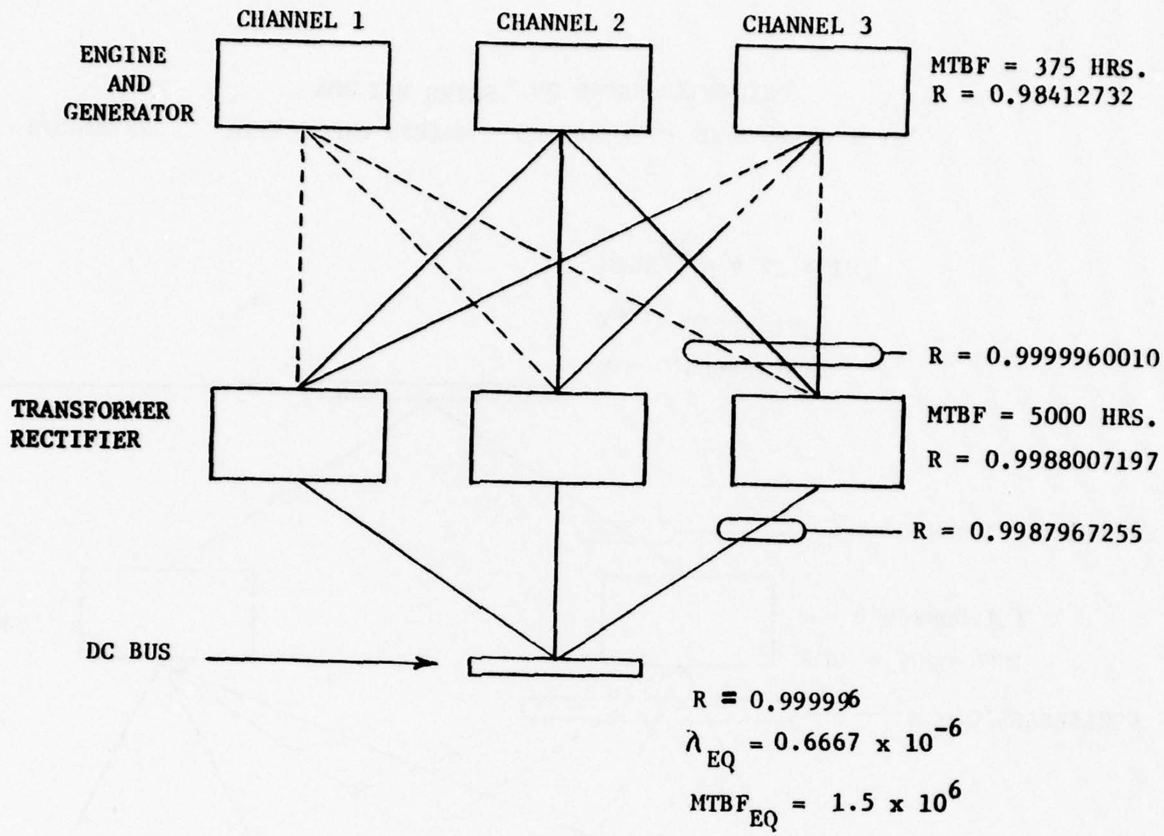


FIGURE 6D MAIN POWER SYSTEM - OPTION 4 ~ THREE GENERATORS, THREE T/R UNITS, AC CROSS-STRAPPING

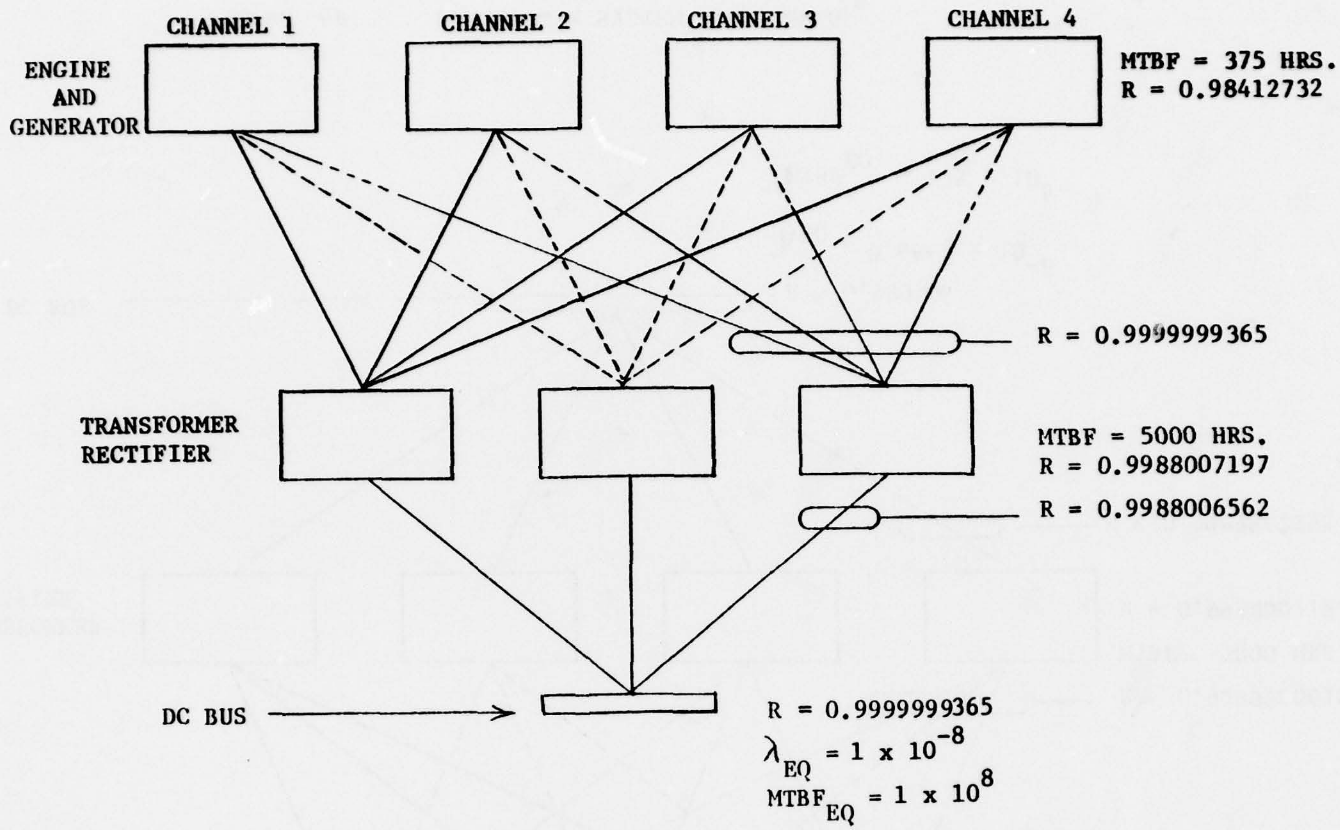


FIGURE 6E MAIN POWER SYSTEM - OPTION 5 - FOUR GENERATORS, THREE T/R UNITS, AC CROSS-STRAPPING

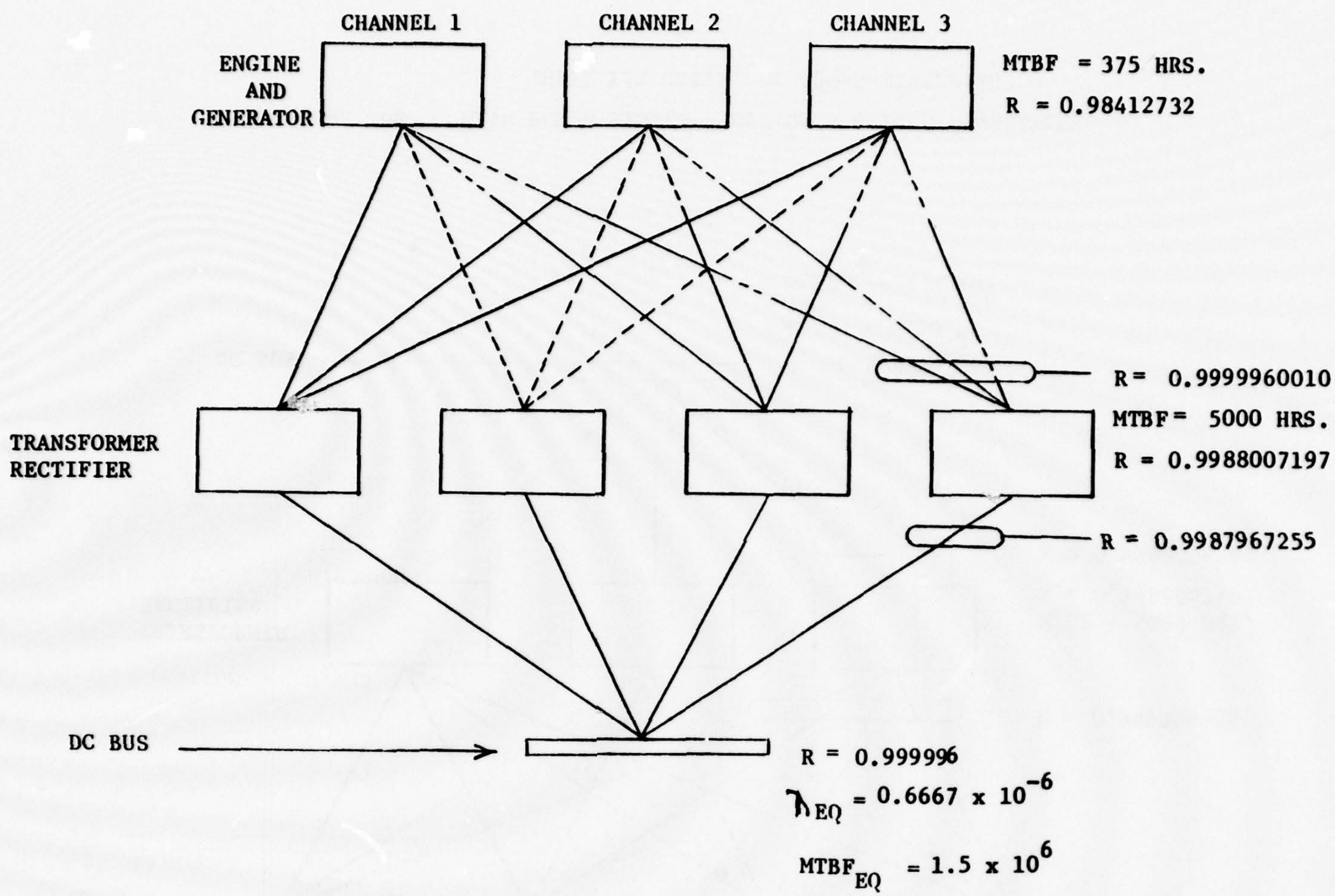


FIGURE 6F MAIN POWER SYSTEM - OPTION 6

THREE GENERATORS, FOUR T/R UNITS, AC CROSS-STRAPPING

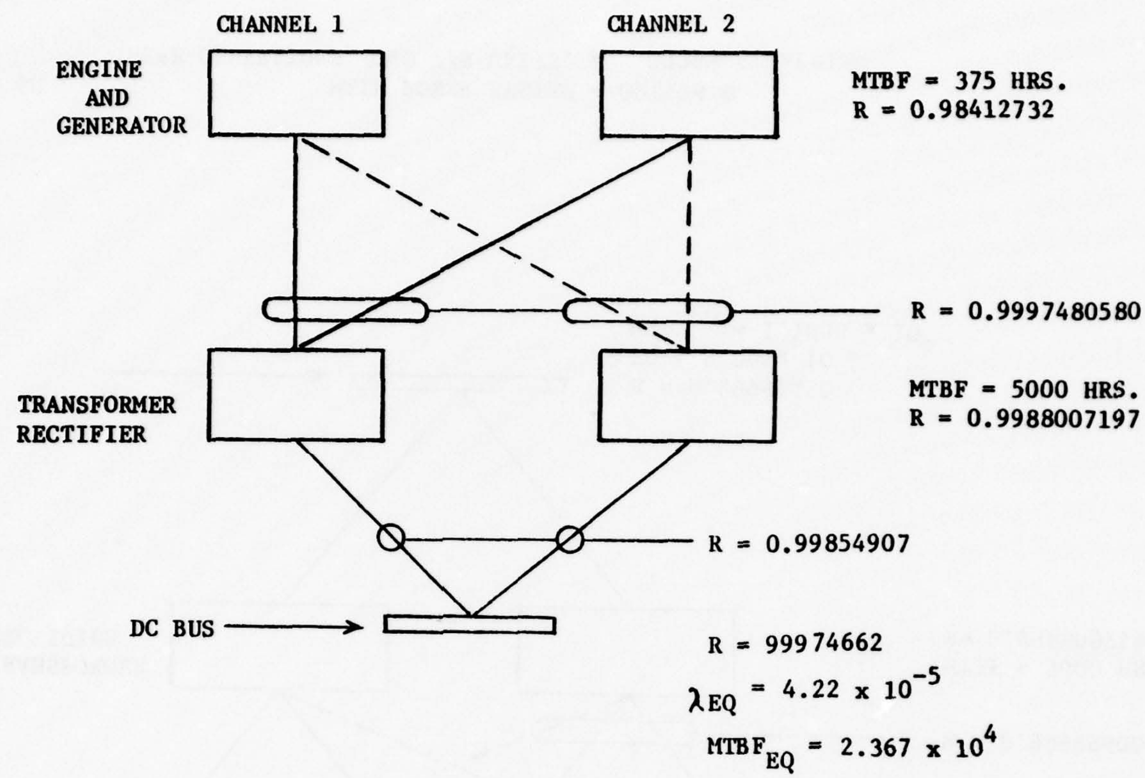


FIGURE 6G MAIN POWER SYSTEM - OPTION 7
TWO GENERATORS, TWO T/R UNITS, AC CROSS-STRAPPING

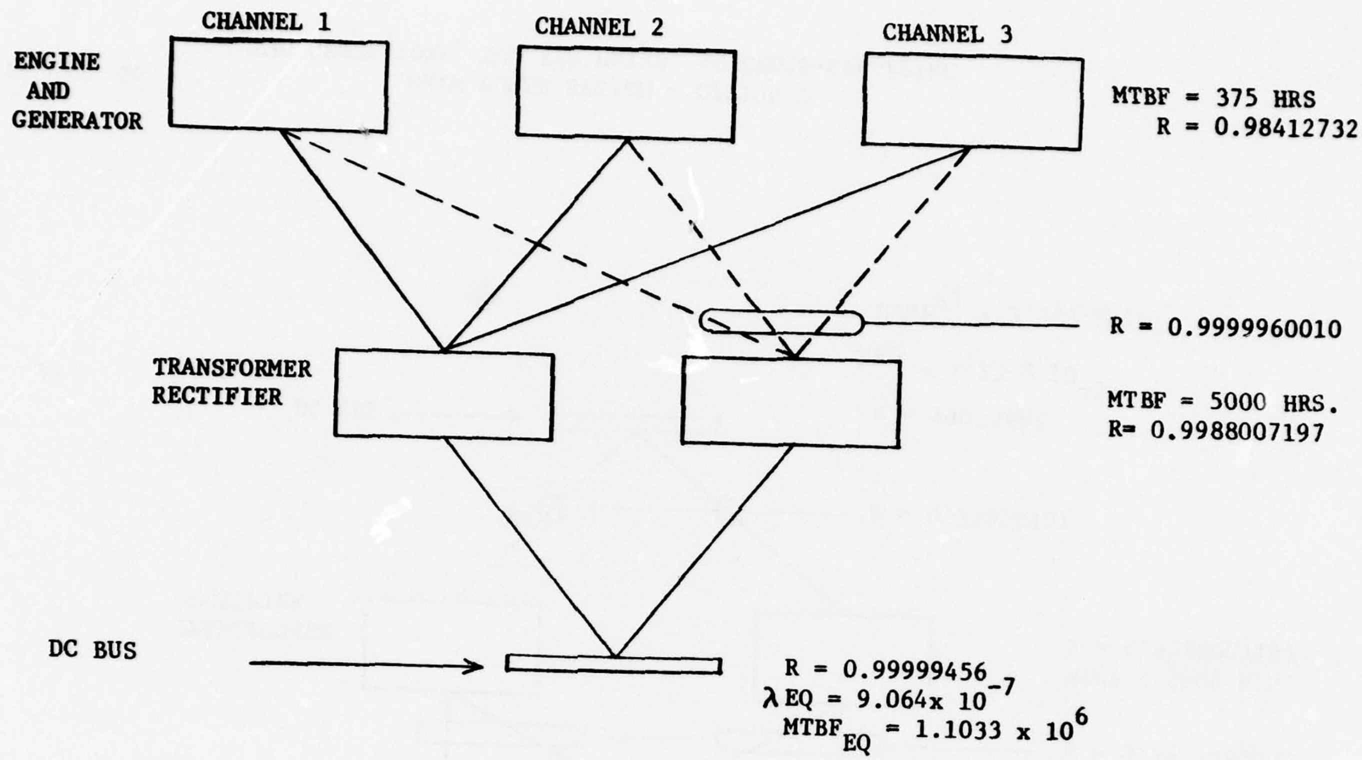


FIGURE 6H

MAIN POWER SYSTEM - OPTION 8
THREE GENERATORS, TWO T/R UNITS, AC CROSS-STRAPPING

for an equivalent primary power source MTBF of 375 hours. The 600 hour MTBF for the turbofan engine represents failures resulting in maintenance action. While very few of these maintenance related failures would result in loss of generator rotation, use of the 600 hour rate yields very conservative reliability predictions. Furthermore, accurate determination of a failure rate from loss of engine rotation was beyond the scope of this study. Tables 4 and 5 summarize the probability of mission success, the effective failure rate and the effective MTBF for the primary and secondary subsystems in the eight multi-engine configuration options. These values are based on a 6 hour mission time. As shown in Table 3, transition from a four generator system to a three generator system results in increasing the failure rate (decreasing reliability) by an approximate factor of 60. However, the reliability of the three channel system is still very high (0.999996). The two generator system suffers a two to three order-of-magnitude increase in failure rate of the four channel system and yields a reliability of 0.999748 which is below the 0.9998 goal established. For this reason, the two channel power system was dropped from further consideration.

Table 5 compares the reliability parameters for various secondary (DC) subsystem options. This table reveals that for a three or four T/R unit system, the DC system reliability is determined virtually by the AC power system reliability and not by the power conversion hardware. When a two T/R unit secondary system is selected, an approximate 50 percent difference in overall power system reliability occurs. This difference becomes more prominent as the primary power system reliability improves. Figure 7 depicts this trend by plotting the DC power bus equivalent MTBF versus generator system MTBF. As shown, at the 1000 hour MTBF used for the generator, a bus MTBF for the two T/R unit system is

TABLE 4

MAIN PRIMARY (AC) POWER SYSTEM
RELIABILITY MATRIX

(INCLUDING ENGINE FAILURES)

OPTION	SYSTEM ARRANGEMENT	RELIABILITY PARAMETERS		
		PROB OF MISSION SUCCESS	EQUIV. (FAILURES/10^6 HOURS)	MTBF EQUIV (HOURS)
1	4 GEN, NO AC CROSS-STRAPPING	0.9999999365	0.01058	9.45×10^7
2, 3, 5	4 GEN, AC CROSS-STRAPPING	0.9999999365	0.01058	9.45×10^7
4, 6, 8	3 GEN, AC CROSS-STRAPPING	0.9999960010	0.6665	1.50×10^6
7	2 GEN, AC CROSS-STRAPPING	0.9997480580	42.	2.38×10^4

TABLE 5

MAIN SECONDARY (DC) POWER SYSTEM RELIABILITY MATRIX

(INCLUDING ENGINE FAILURES)

OPTION	SYSTEM ARRANGEMENT	RELIABILITY PARAMETERS		
		PROB OF MISSION SUCCESS	λ EQUIV. (FAILURES/10^6 HRS.)	MTBF EQUIV (HOURS)
1	4 GEN, 4 T/R UNITS NO AC CROSS-STRAPPING	0.9999999154	.01409	7.095×10^7
2	4 GEN, 4 T/R UNITS AC CROSS-STRAPPING	0.9999999365	0.01	1×10^8
3	4 GEN, TWO T/R UNITS CROSS-STRAPPING	0.9999985615	0.24	4.17×10^6
4	3 GEN, 3 T/R UNITS CROSS-STRAPPING	0.999996	0.6667	1.5×10^6
5	4 GEN, 3 T/R UNITS CROSS-STRAPPING	0.9999999365	0.01	1.0×10^8
6	3 GEN, 4 T/R UNITS CROSS-STRAPPING	0.999996	0.6667	1.5×10^6
7	2 GEN, 2 T/R UNITS CROSS-STRAPPING	0.99974662	42.2	2.367×10^4
8	3 GEN, 2 T/R UNITS CROSS-STRAPPING	0.99999456	0.906	1.1×10^6

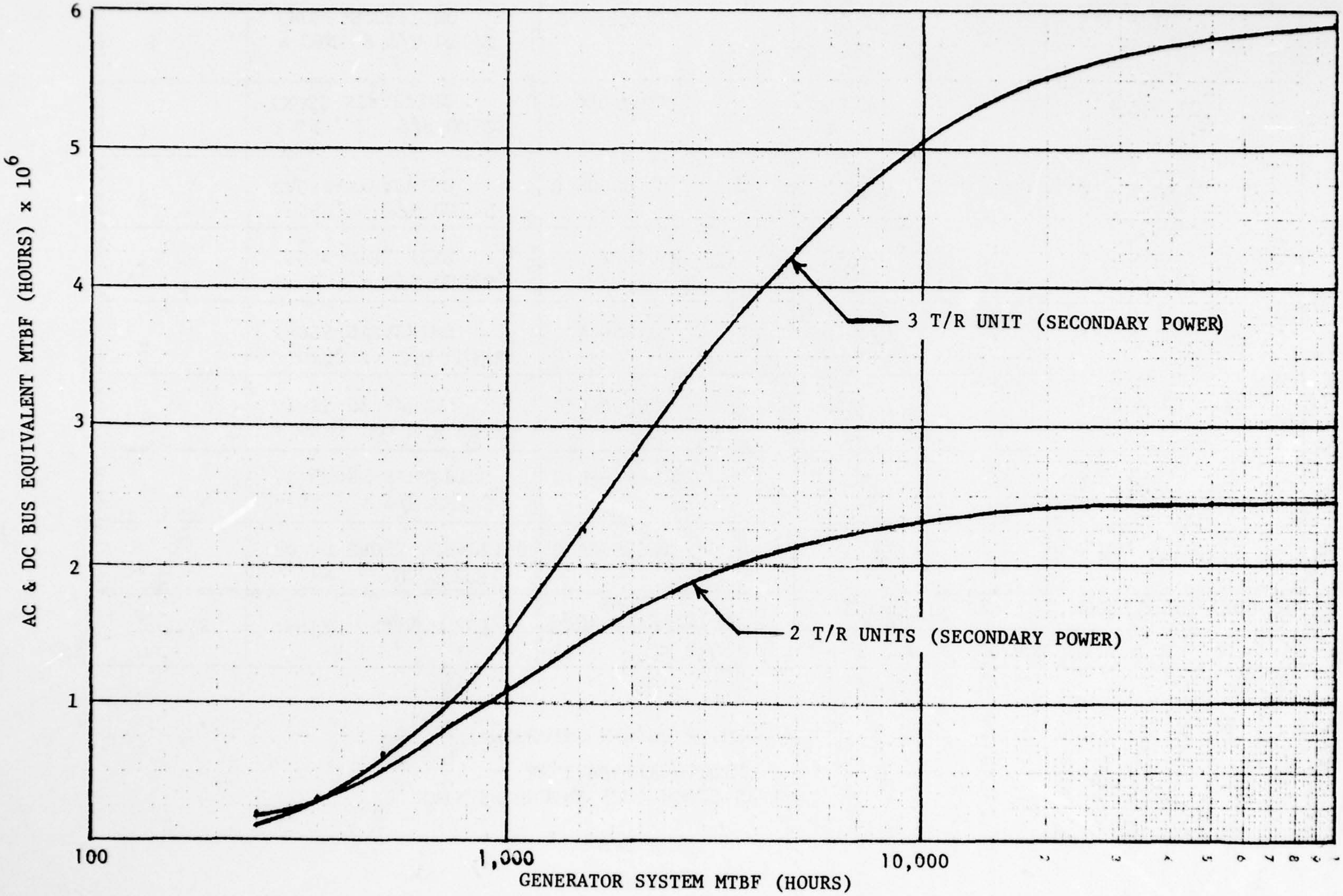


FIGURE 7 POWER BUS MTBF VS GENERATING SYSTEM MTBF

approximately 70 percent of the three T/R system MTBF. At 10,000 hour generator MTBF, the two T/R system MTBF decreases to 45 percent of the three T/R system. A similar relationship of increasing degradation would occur as the engine MTBF is increased from the 600 hour assumed base.

To maximize the improvement in bus reliability due to primary power system reliability improvement, a three T/R power conversion system is preferred over the dual T/R system.

Complete elimination of the secondary power buses should, however, be pursued.

Based solely on reliability trades, the main power system shown in Figure 8 is the best for the four engine aircraft and easily meets the nominal mission completion probability of 0.9998. Significantly higher or lower mission completion requirements from those assumed will influence the recommended configuration for any specific aircraft application. It is noted that the mission completion probability of 0.999996 is conservative in comparison with other critical aircraft subsystems such as fuel transfer, hydraulics, propulsion, structures, etc. Finally, the number of channels used on a four engine aircraft is influenced by factors other than reliability. The reliability requirement, however, does establish the lower limit on generator quantities. The upper limit on generator quantities is influenced by such factors as:

- o Aircraft weight and balance restrictions
- o Available istallation space

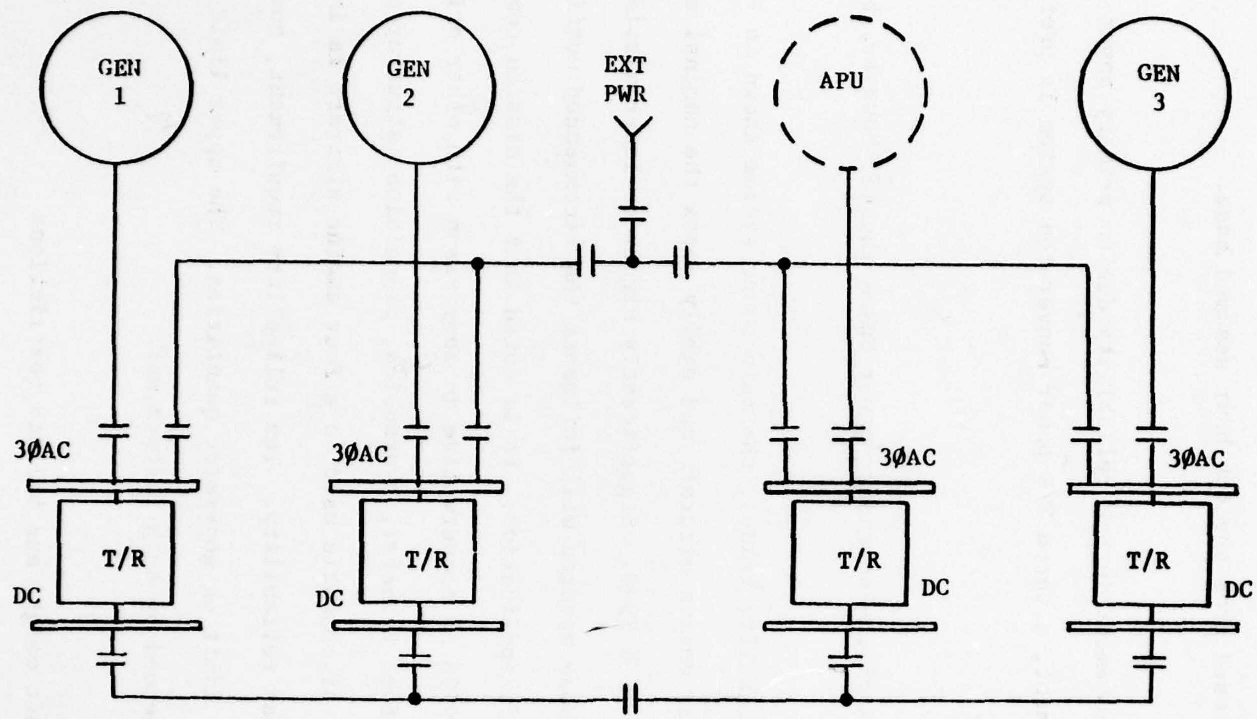


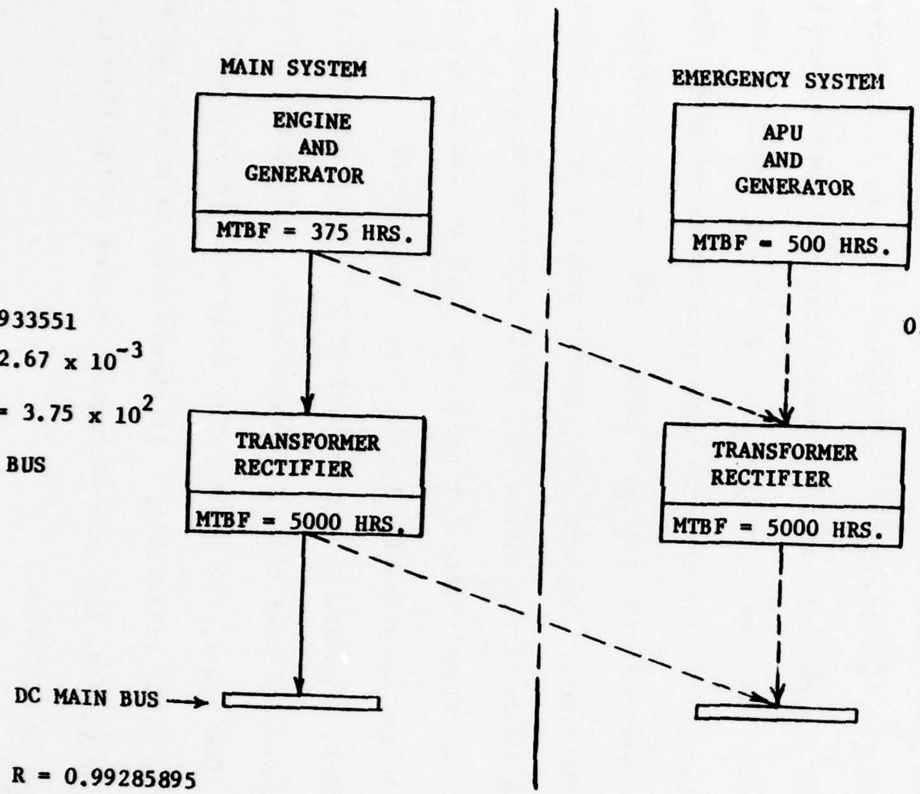
FIGURE 8 BEST MAIN POWER SYSTEM CONFIGURATION BASED ON RELIABILITY ASSESSMENT

- Type of engine starting system (electric start may imply starter/generator on each engine)
- Electric load

For these reasons, any optimum generating system is subject to change with each specific aircraft application or mission.

A reliability assessment was also made on a single engine power generation system. Since only one main prime mover (engine) is available in this aircraft class, the number of generator system configurations is minimal. Figure 9 illustrates a reliability block diagram for the only reasonable hardware configuration. This diagram depicts the system interrelationships for both the main and emergency power sources. The top reliability block in each of the two subsystems contains the generator hardware plus the prime mover. The prime mover for the main system is the aircraft engine. This prime mover has a predicted MTBF of 600 hours. The emergency system prime mover is an APU with a 1000 hour MTBF. As shown in the figure, the probability of mission success (main generating system) is slightly short of the 0.995 goal defined in paragraph 3.1. This low reliability is due primarily to the "low" engine MTBF of 600 hours. With this engine MTBF, a generator system MTBF of 7228 hours minimum (likely attainable) is required to meet the 0.995 requirement. Likewise, with a generator MTBF of 1000 hours, the engine MTBF would need to be greater than 1242 hours to provide the 0.995 reliability. It should also be noted that with a 1000 hour MTBF generating system, the main electrical system reliability is 0.997 when the engine failures are excluded.

As also illustrated in Figure 9, the probability of safe return (emergency system) is well within the 0.9998 goal with an estimated reliability of 0.99998039.



$R = 0.9933551$

$\lambda_{EQ} = 2.67 \times 10^{-3}$

$MTBF_{EQ} = 3.75 \times 10^{2}$

AC MAIN BUS

0.99750312

AC EMERG BUS

$R = 0.99998341$

$R = R_{MAIN\ AC} + (R_{EMERG\ AC})(1 - R_{MAIN\ AC})$

$\lambda_{EQ} = 6.636 \times 10^{-6}$

$MTBF_{EQ} = 1.50685 \times 10^{5}$

$R = 0.99285895$

$\lambda_{EQ} = 2.867 \times 10^{-3}$

$MTBF_{EQ} = 3.488 \times 10^{2}$

MISSION TIME = 2.5 HRS.

DC EMERG BUS (if required)

$R = R_{MAIN\ DC} + (R_{EMERG\ DC})(1 - R_{MAIN\ DC})$

$R = 0.99998039$

$\lambda_{EQ} = 1.569 \times 10^{-5}$

$MTBF_{EQ} = 6.37 \times 10^{4}$

MISSION TIME = 1.25 HRS FOR EMERG COMPONENTS AND 2.5 HRS. FOR MAIN COMPONENTS.

FIGURE 9 MAIN AND EMERGENCY POWER SYSTEM - SINGLE ENGINE AIRCRAFT

5.2.2 POWER TO UTILIZATION EQUIPMENT

In order to establish the reliability for delivering power to the utilization equipment, the level of reliability degradation imposed by the power distribution system needs to be determined. This degradation, however, is a direct function of the number of systems or circuits required for aircraft safe return and mission completion. A simplified estimate of this degradation is derived by assuming a linear relationship between power distribution failure rate and the number of circuits required. This simplification is reasonable if any failures from common hardware such as EMUX processors, data buses, etc., are accounted for at the bus management level. The linear estimate also assumes that an "average" power distribution circuit can be defined to determine a representative failure rate per circuit. Figure 10 illustrates reliability block diagrams for representative conventional and advanced technology (solid state) power distribution circuits. Figure 11 is a plot of power distribution system reliability versus the quantity of circuits. Estimated circuit quantities required for aircraft safe return and mission completion circuits for a single engine and a multiple (four) engine aircraft are identified on the figure. The point estimates yield the reliability requirements for the power distribution systems shown in Table 6. The probability of delivering power to loads can now be calculated by combining the reliability of the distribution system with the reliability of power at the buses. These values are shown in Table 6. It should be emphasized that the reliability values shown in Table 6 apply to supplying power to all the loads. The table also shows the improved reliability of the advanced technology power distribution system over the conventional system, especially when mission time is long as is typical for multi-engine aircraft.

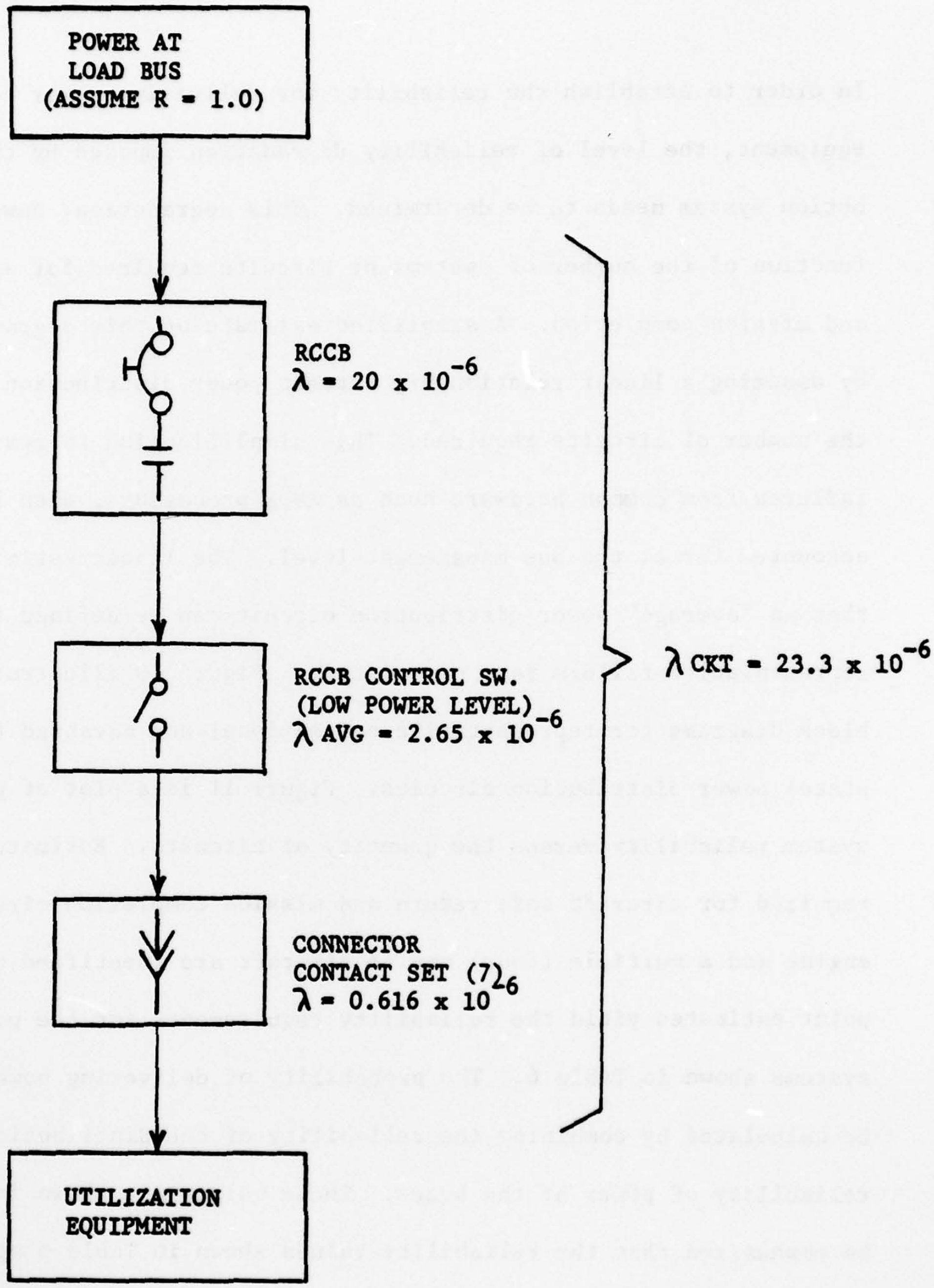


FIGURE 10A RELIABILITY DIAGRAM OF TYPICAL CONVENTIONAL POWER DISTRIBUTION SYSTEM

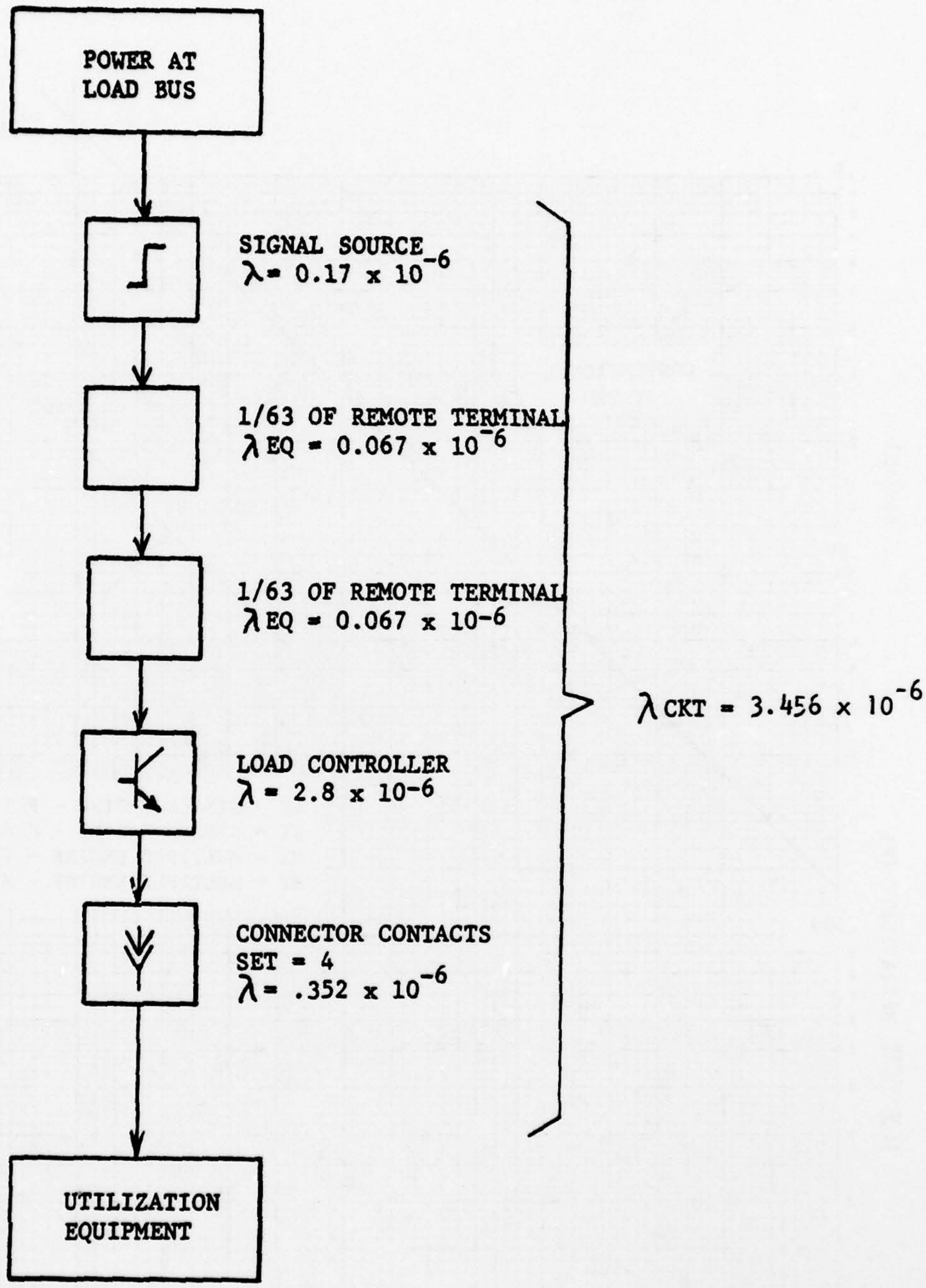


FIGURE 10B RELIABILITY DIAGRAM OF TYPICAL ADVANCED TECHNOLOGY POWER DISTRIBUTION CIRCUIT

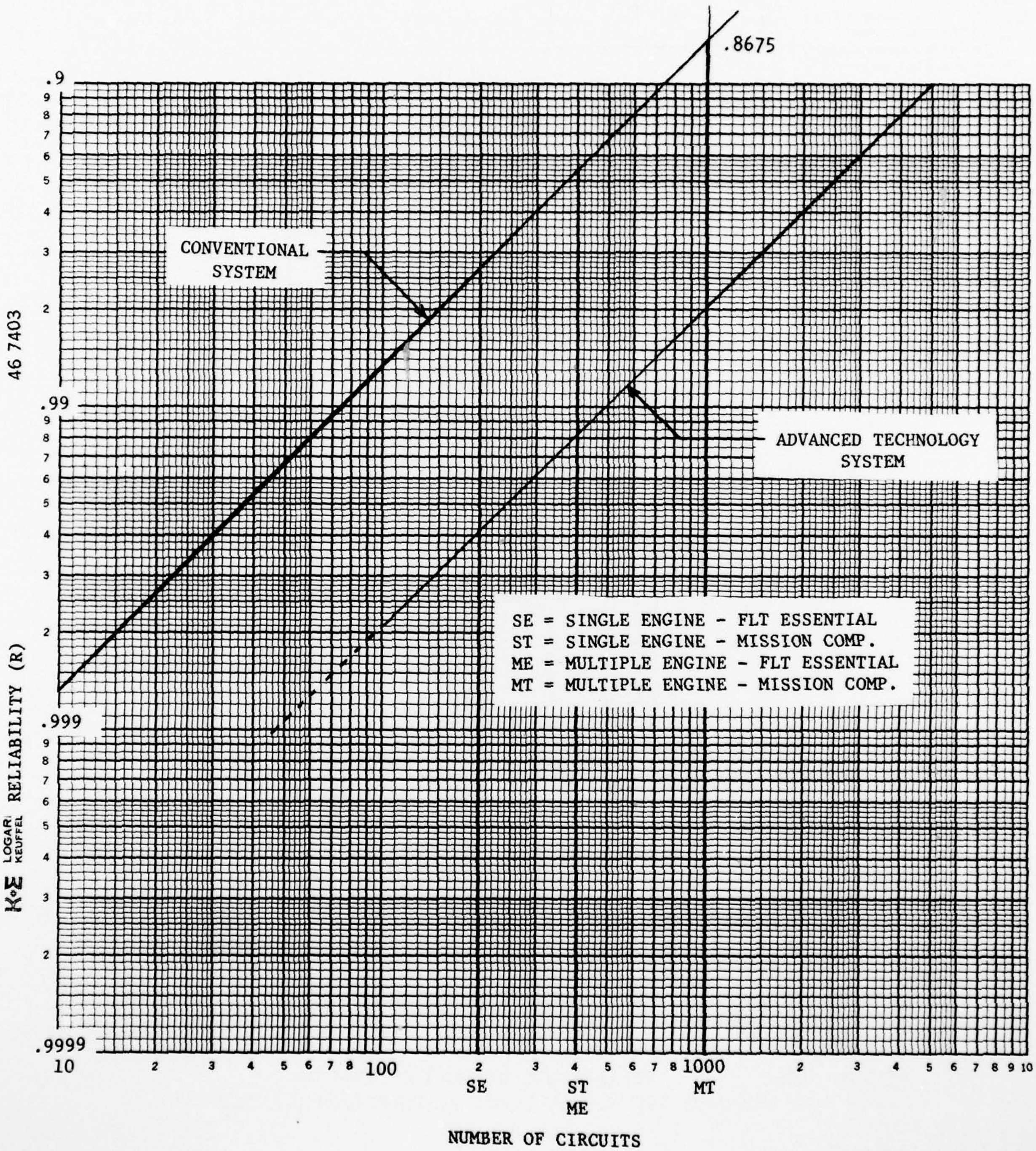


FIGURE 11 POWER DISTRIBUTION SYSTEM RELIABILITY

TABLE 6

POWER SYSTEM RELIABILITY REQUIREMENTS SUMMARY

SYSTEM CONFIGURATION		POWER AT BUSSES (A)	POWER DISTRIBUTION (B)		POWER TO ALL LOADS (C)	
			CONVENTIONAL	ADV. TECH	CONVENTIONAL	ADV. TECH.
SINGLE ENGINE (OPERATE TIME 2.5 HOURS)	SAFE RECOVERY	0.9998	0.9893	0.9984	0.9891	0.9982
	MISSION COMPLETION	0.995	0.9788	0.9967	0.9739	0.9918
MULTIPLE ENGINE (OPERATE TIME 6 HOURS)	SAFE RECOVERY	0.999995	0.9456	0.9915	0.9455	0.9913
	MISSION COMPLETION	0.9998	0.8675	0.9790	0.8673	0.9788

C = A x B

5.2.3 FLY-BY-WIRE SYSTEMS

Fly-by-wire aircraft depend on the electrical power system for flight control which requires higher reliability and much faster transfer to back-up power. In general, a power system for a fly-by-wire aircraft consists of:

- Main generator(s) installed on the engine pad with the generator(s) rated for total electrical load.
- An APU driven generator for emergency power.
- A battery power source for the time period between main power loss and start-up of the APU system or for added reliability through redundancy.

A gross analysis of the general level of power system reliability that can be expected from this arrangement for a single engine aircraft is shown in Figure 12. This analysis assumes ballpark component MTBF's of 1000 hours for the generator (including GCU's) and the APU, and an estimated 100,000 hour MTBF for in-flight loss of engine rotation. In addition, a 2.5 hour mission time is assumed for the single engine system. As shown on the figure, electrical power should be available with a probability of 0.99999998. This probability does not account for a momentary power loss which will occur during switchover from main to the APU in the event of a main power loss. The purpose of the battery power source is to maintain power during this switchover period. The probability that power is available either from the main source or from the battery/inverter source is 0.99999983 as shown in the figure. This probability will reduce if the APU start-up time is extended or if the APU fails to start.

Equating these probabilities to FBW requirements results in the following:

- MIL-F-9490 requires a maximum aircraft loss rate due to flight control system (or supporting subsystem) failures of 100×10^{-7} losses per flight hour for MIL-F-8785 Class I, II or IV aircraft. (Most single engine aircraft are Class IV.) This loss rate equates to an equivalent MTBF of 100,000 hours.
- If one assumes that the power system reliability should be at least an order of magnitude higher than the flight control system, then an electrical power equivalent MTBF of 1,000,000 hours is required.
- For the 2.5 hour defined mission time, the 10^6 hour MTBF equates to a power system reliability requirement of 0.9999975. This level is within the 0.999999 probability derived for the system configuration shown in Figure 12.

The FBW reliability requirements for Class III (heavy bombers) aircraft are more stringent. For these multi-engine aircraft, a loss rate of less than 5×10^{-7} is required which equates to an equivalent MTBF of 2,000,000 hours. The electrical power for these aircraft requires a 2×10^7 hour equivalent MTBF or a reliability of 0.9999997 for a 6 hour mission. This level is attainable with a four channel system as can be seen in Table 4.

Another technique for improving the reliability and assuring "no-gap" power is shown in Figure 13. Basically, redundant power distribution circuits are provided from two or more LMC's, depending on the reliability required. The FBW equipment contains an equal number of power supplies with their outputs

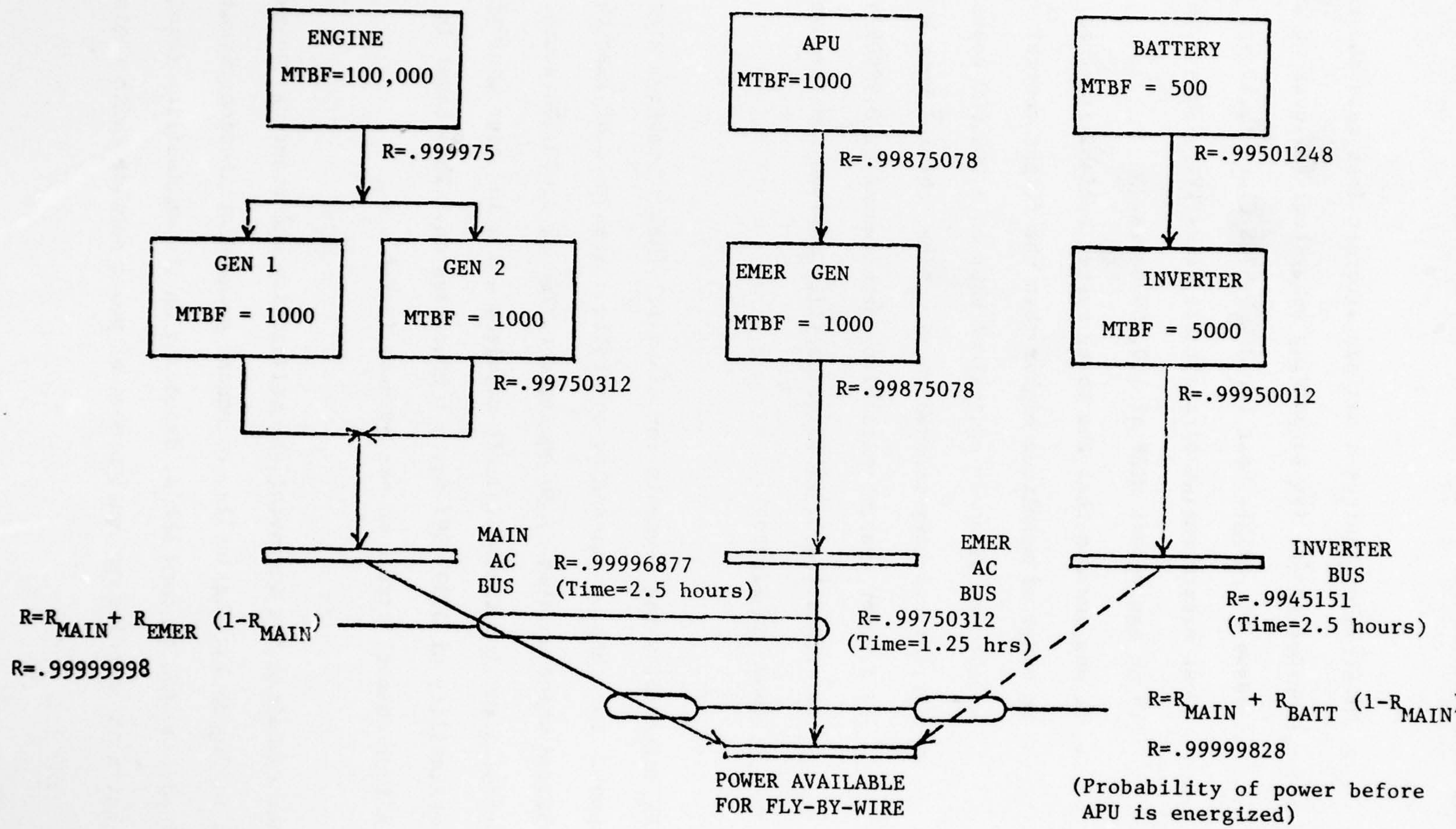


FIGURE 12 RELIABILITY BLOCK DIAGRAM FOR SINGLE ENGINE FLY-BY-WIRE CONFIGURATION

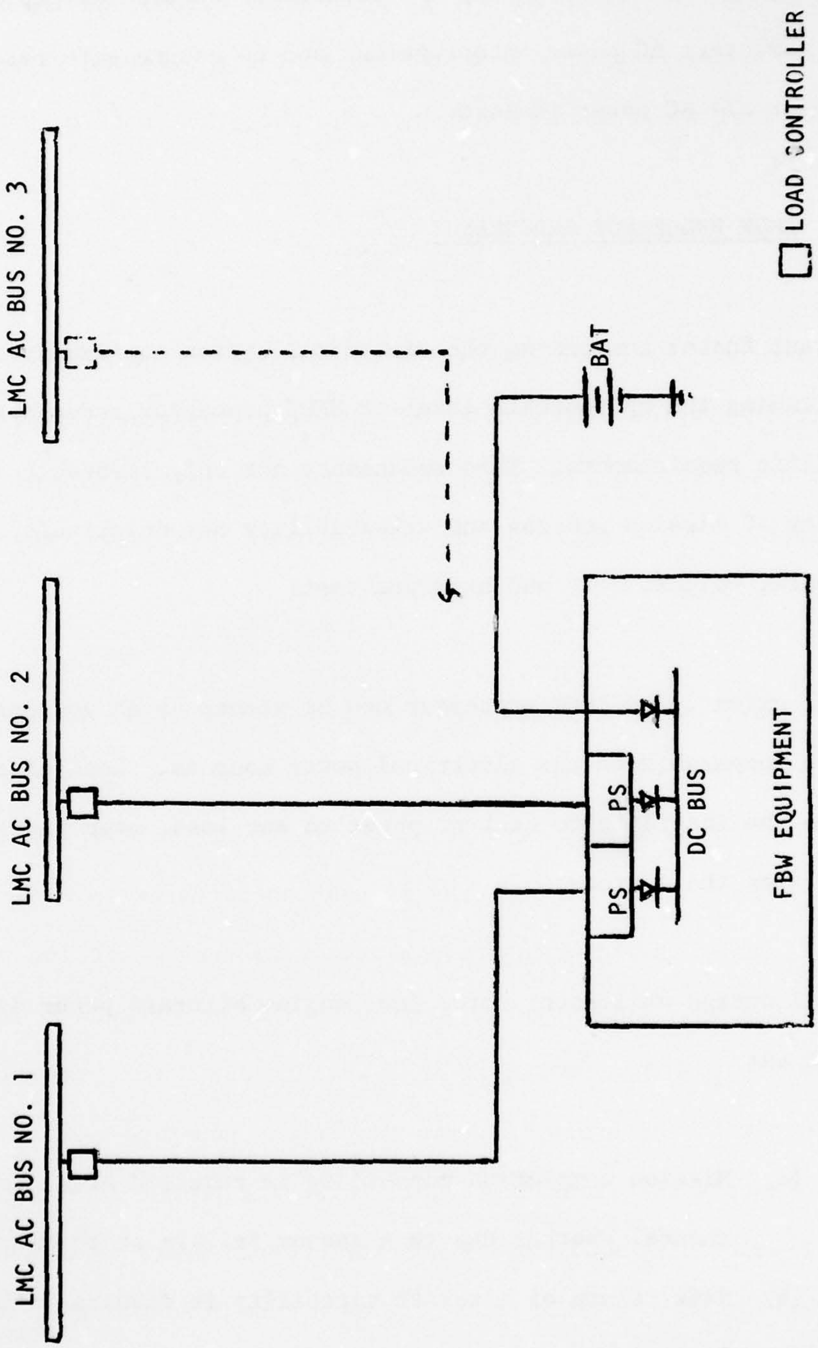


FIGURE 13 POWER DISTRIBUTION CIRCUIT REDUNDANCY

AD-A078 629 VOUGHT CORP DALLAS TX F/G 1/3
POWER SYSTEM CONTROL STUDY. PHASE I - INTEGRATED CONTROL TECHNI--ETC(U)
JUN 79 D E LAUTNER , A J MAREK , J R PERKINS F33615-78-C-2018
UNCLASSIFIED AFAPL-TR-79-2084 NL

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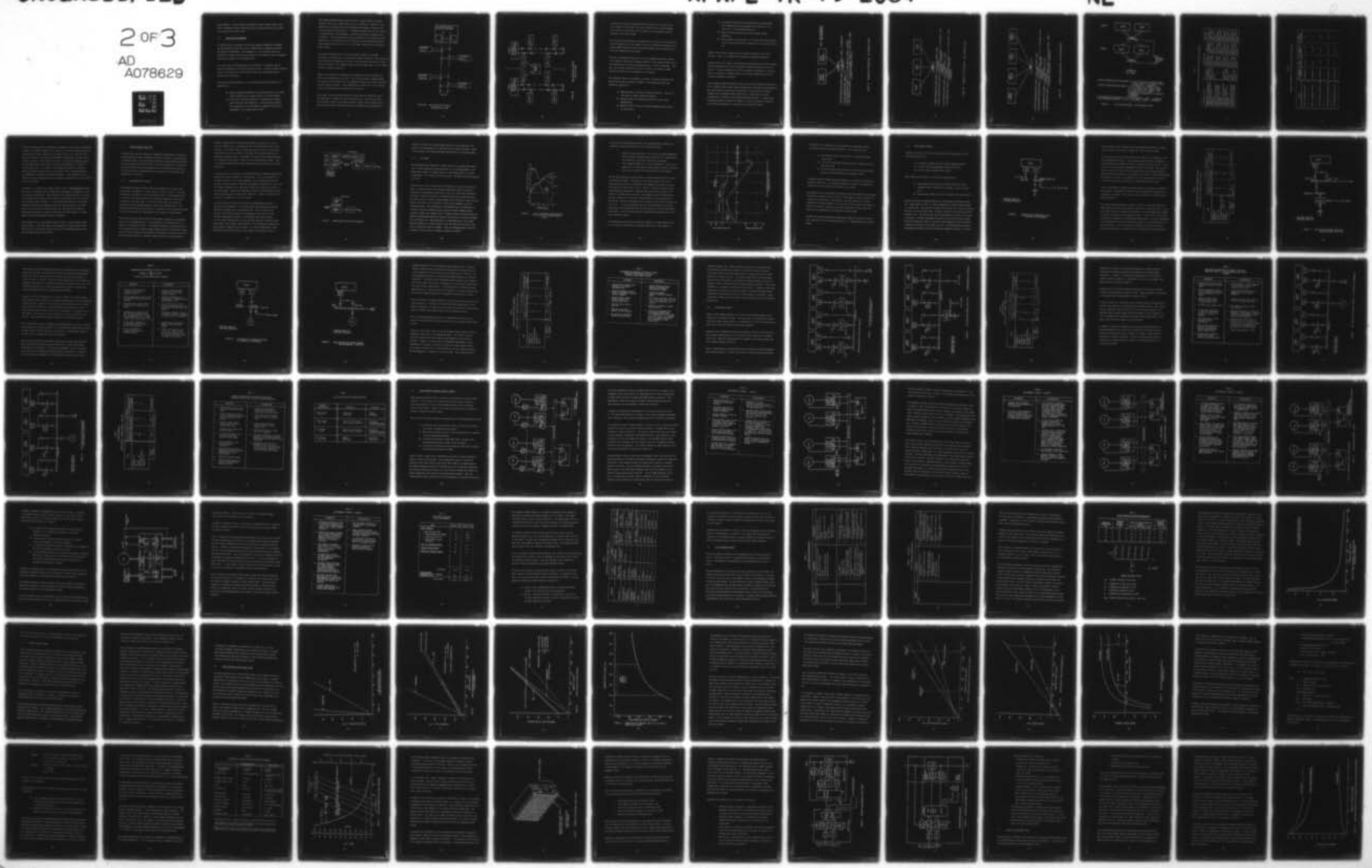
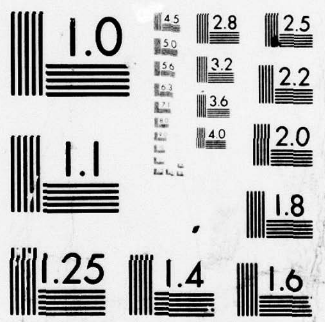




MICROCOPY RESOLUTION TEST CHART
NATIONAL BUREAU OF STANDARDS-1963-A

diode isolated. A battery supply is provided to assure "no-gap" power in the event of momentary AC power interruptions and to assure safe return of aircraft in the event all AC power is lost.

5.3 EMUX PROCESSOR REDUNDANCY

An important factor concerning the electrical system configuration and design is establishing the appropriate level of EMUX processor redundancy required to meet specific requirements. This redundancy not only favorably impacts the probability of mission success and vulnerability but negatively impact maintenance failure rate, weight, size and hardware cost.

To a large extent, the EMUX processor can be viewed as an equipment item of importance comparable to the electrical power sources. Loss of all EMUX processors results in the inability to deliver power to any load, even though power is available from the sources.

The general design philosophy for a four engine aircraft power system can be summarized as:

(a) Mission completion capability is required after loss of one power channel whether due to a random failure or to battle damage.

(b) Safe return of aircraft capability is required after complete loss of the main power system (i.e., all main power channels). This capability is to be provided by an emergency power source independent of the primary power sources.

This design philosophy implies operation under a scenario wherein one power channel is lost due to random failure prior to reaching the target area. The mission can be continued due to the availability of adequate, reliable power from the other main power sources. If additional channels are lost as a result of random failure or battle-damage, a decision on aborting the mission would be required. This decision would typically be based on the criticality of the mission and on the reliability and vulnerability of the remaining operational power channel(s) (none, one, two, etc., main power channels plus the emergency power channel).

By applying this same scenario and system design philosophy to the EMUX processors, an architecture is evolved which requires a minimum of three processors. This architecture accommodates two failures with sufficient capability to safely return the aircraft. Two candidate EMUX architectures applicable to a four engine aircraft are shown in Figure 14.

Figure 14A illustrates an integrated bus arrangement where each processor can control all aircraft electrical loads. This approach permits a minimum number of processors to provide the level of reliability required for mission completion and safe return of aircraft. This minimization is possible due to the complete level of operational backup provided by each processor.

In contrast, the split bus system shown in Figure 14B will inherently require more processors to achieve the same reliability level. Since each processor only controls a portion of the total electrical system, each processor does not provide the full system backup provided by the integrated bus implementation.

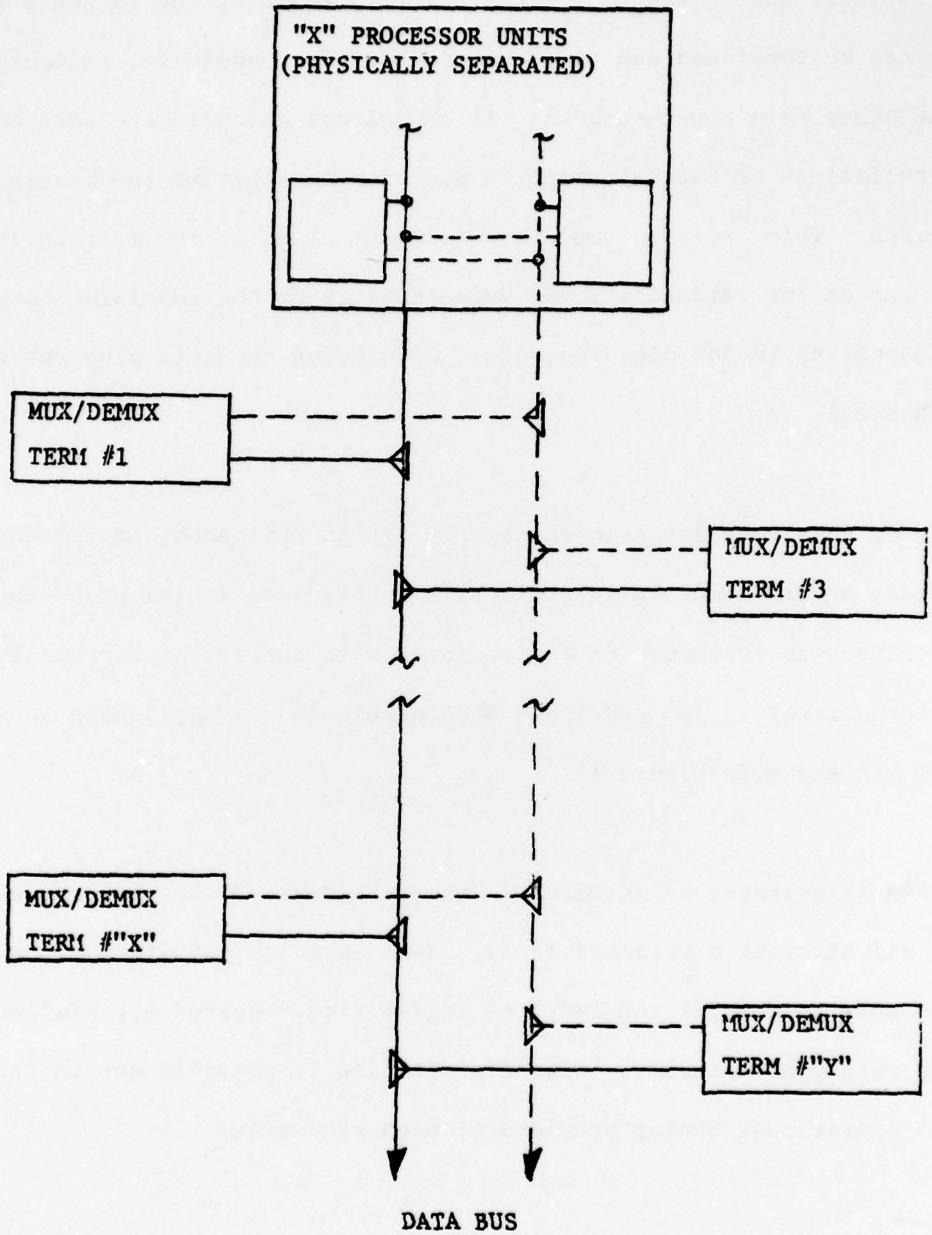


FIGURE 14A EMUX ARCHITECTURE OVERVIEW
INTEGRATED BUS SYSTEM

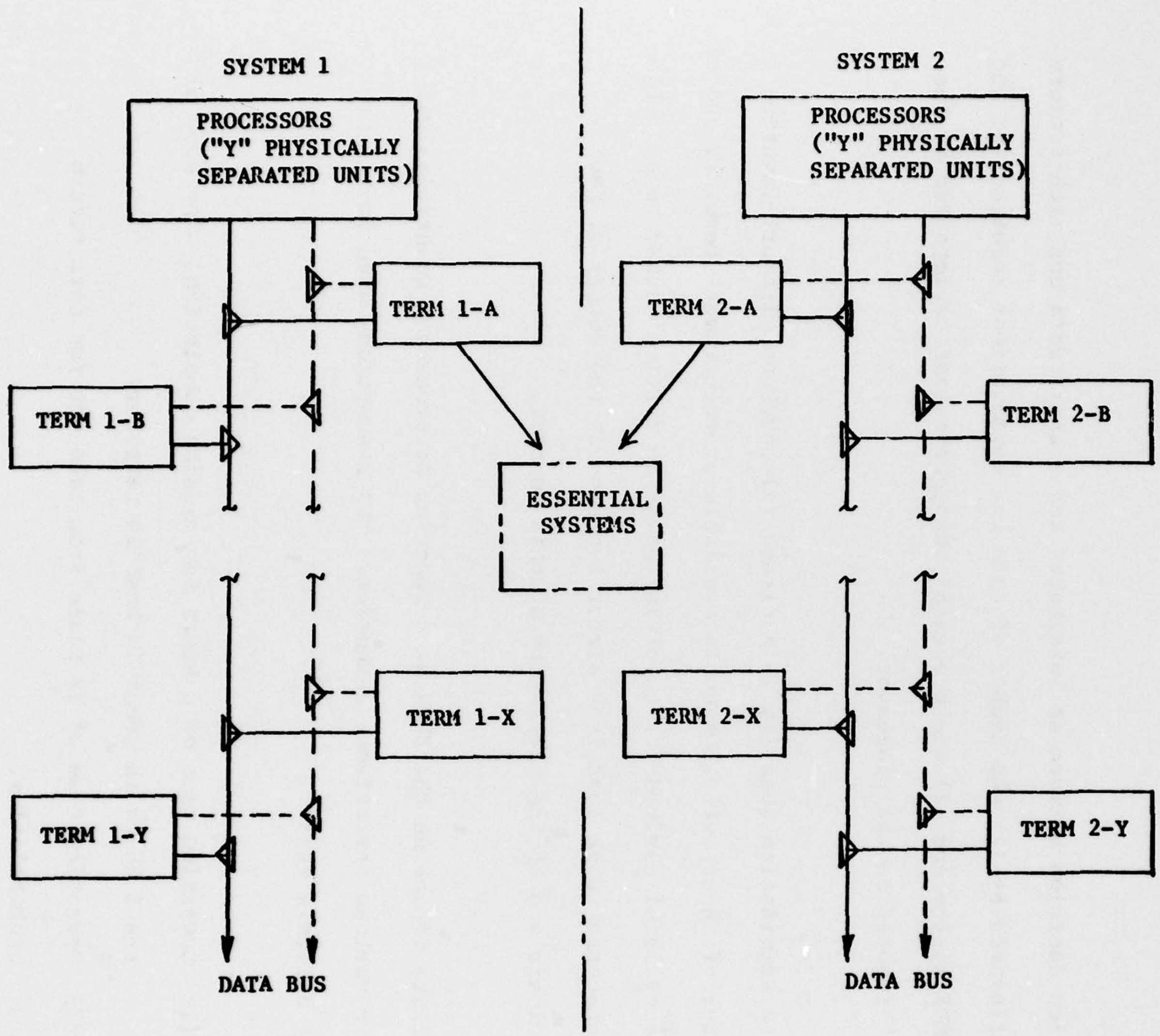


FIGURE 14B EMUX ARCHITECTURE OVERVIEW
SPLIT BUS SYSTEM

100

To minimize the need for adding additional processors to the split system and to improve the processor "safe return" failure rates, flight essential loads critical to safe return of the aircraft can be redundantly serviced from each of the split systems.

The selection decision between an integrated and a split data bus architecture is also influenced by (1) the number of data bus terminations required for the complete EMUX system and (2) the processor throughput time as impacted by the "work load" imposed on the processor.

The data bus termination quantity is a direct (although non-linear) function of the number of input/output channels available at each remote terminal and of the redundancy level desired for supplying power to critical subsystems. Likewise, the processor work load is a direct function of the number of EMUX controlled loads and of the complexity of this control.

The reliability impact on the EMUX as a function of processor quantities was analyzed for various operational scenarios. The assumptions used in the reliability analysis are:

(1) Operation time of 6 hours for mission completion. One hour of the 6 hours is spent in hostile territory.
(2) Operation time of 12 hours from take-off for safe return probabilities.
(3) Operation time of 6 hours for mission completion plus 6 hours for safe return.

(4) The effective failure rate of one processor providing proper data communication and processing over at least one of two data buses is 100 failures/10^6 per hour.

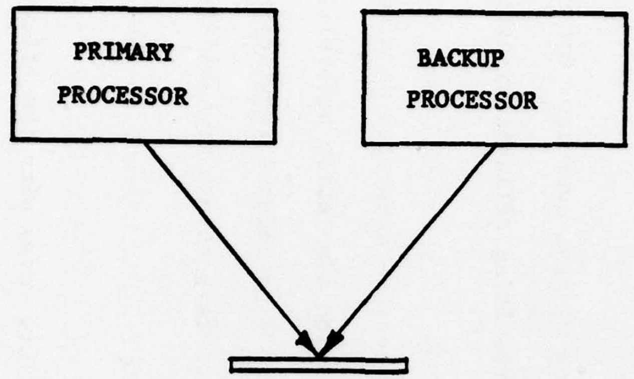
(5) Mission is aborted whenever only one processor remains operational.

(6) Battle damage to the processor, if it occurs, will fail processor as soon as hostile territory is entered (i.e., 5 hours after take-off).

Figure 15 and 16 depict reliability block diagram overviews for the cases analyzed. Table 7 is a summary of the resulting reliability parameters.

As can be seen in the table, the two processor-integrated data bus system (Figure 14A) is unacceptable due to the zero mission completion probability after one failure. This zero probability is due to a procedural requirement to abort the mission to provide an acceptable level of safe return probability and not due to actual reliability predictions.

Table 8 compares EMUX bus/processor failure rates with the effective failure rate of two candidate main AC power system configurations. Assuming the three generator main power system is selected as the baseline, the three and four processor-integrated bus arrangements have, respectively, a three and four order of magnitude better reliability than the power generation system. A single order of magnitude higher reliability for EMUX is all that is necessary to eliminate any significant degradation of the overall power generation system.



λ_{EFF} = 100 FAILURES/10^6 HRS PER PROCESSOR

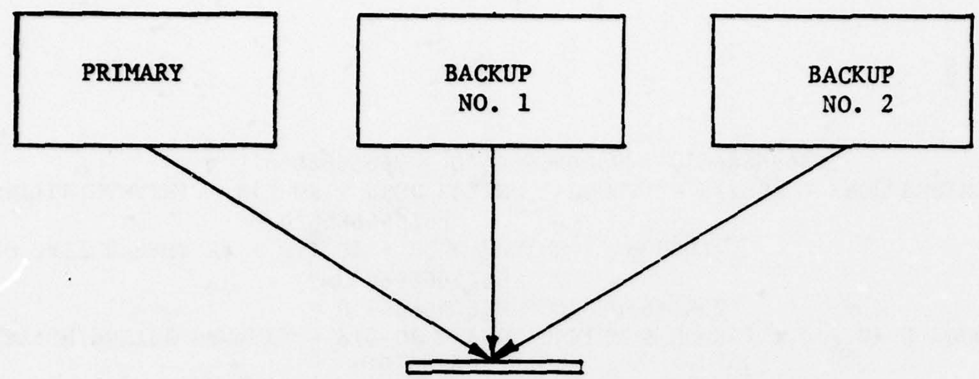
P (MISSION COMPLETION/NO BATTLE DAMAGE) = P(2 PROC DURING 6 HOURS) = $(0.9994001799)^2$
= 0.9988007197

P (MISSION COMPLETION/BATTLE DAMAGE) = 0.0

P (SAFE RETURN/NO BATTLE DAMAGE) = P(1 OF 2 PROC DURING 12 HOURS) = 0.99999856

P (SAFE RETURN/BATTLE DAMAGE) = P(1 OF 2 PROC DURING 5 HOURS) x P(1 PROC DURING 7 HOURS)
= 0.99999975 x 0.99930025 = 0.9993

FIGURE 15A INTEGRATED BUS ARCHITECTURE - TWO PROCESSOR SYSTEM



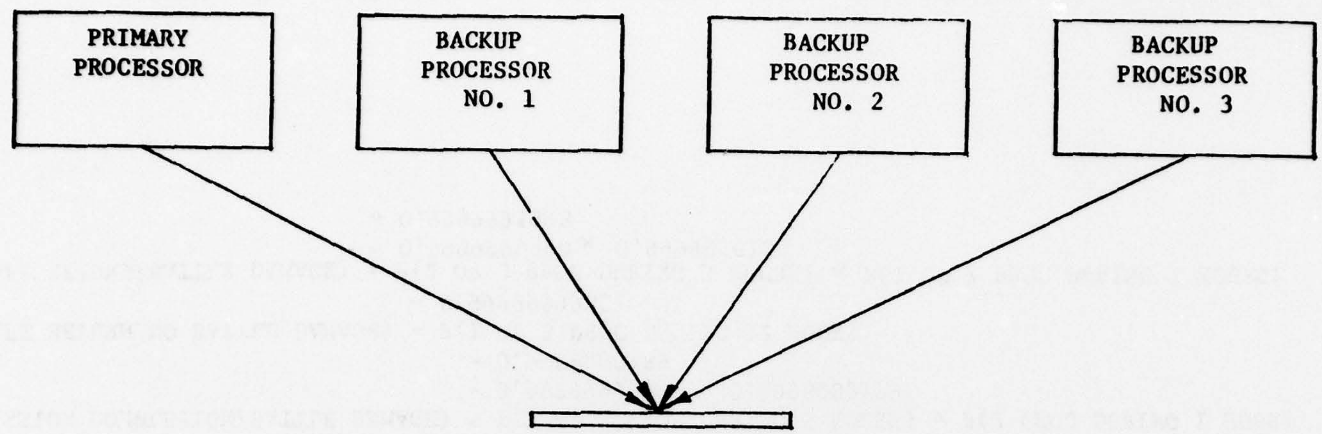
P (MISSION COMPLETION/NO BATTLE DAMAGE) = P(2 OF 3 PROC DURING 6 HOURS)
= 0.9999999982

P (MISSION COMPLETION/BATTLE DAMAGE) = P(2 OF 3 PROC DURING 5 HOURS) x P(2 PROC DURING 1 HOUR)
= 0.99999999990 x 0.9998000199
= 0.9998000189

P (SAFE RETURN/NO BATTLE DAMAGE) = P(1 OF 3 PROC DURING 12 HOURS)
= 0.9999999982

P (SAFE RETURN/BATTLE DAMAGE) = P(2 OF 3 PROC DURING 5 HOURS) x P(1 OF 2 PROC DURING 7 HOURS)
= 0.9999999990 x 0.9999995103
= 0.9999995093

FIGURE 15B INTEGRATED BUS ARCHITECTURE - THREE PROCESSOR SYSTEM



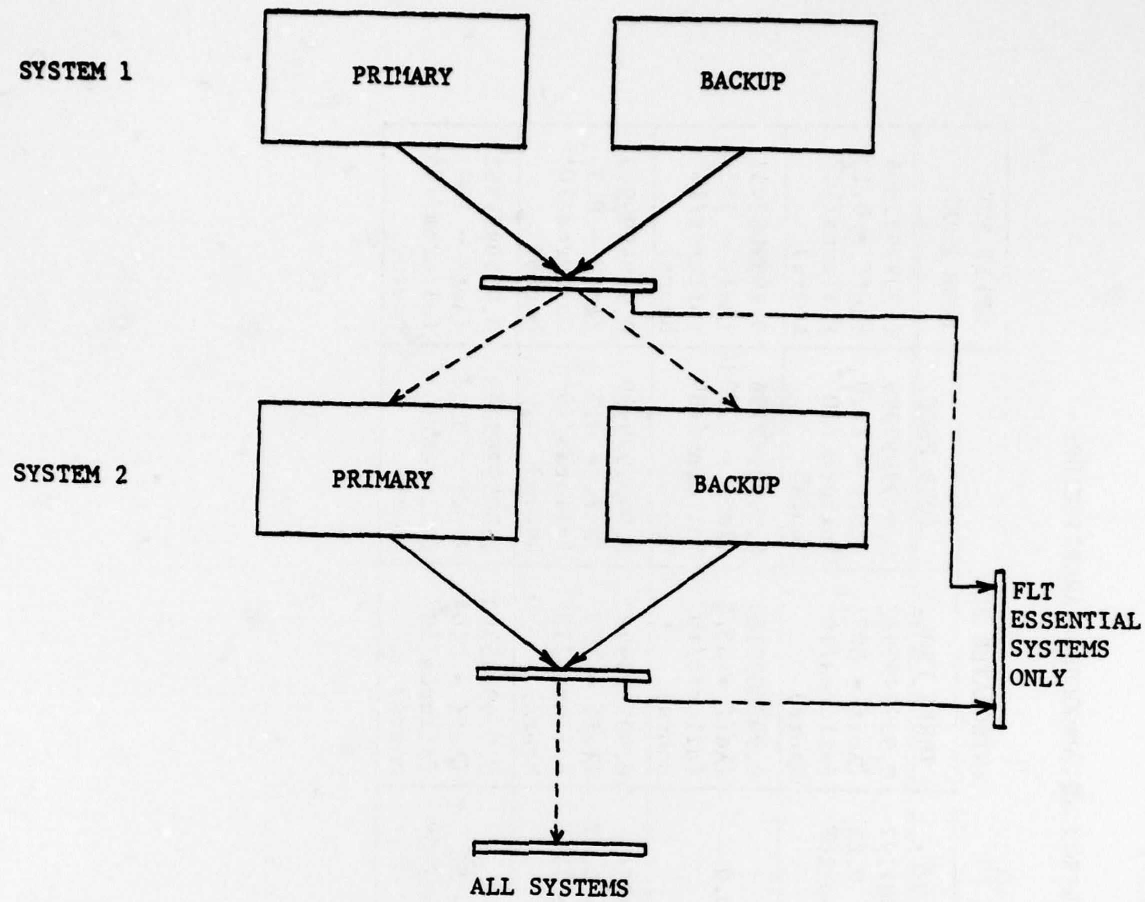
P (MISSION COMPLETION/NO BATTLE DAMAGE) = P(2 OF 4 PROC DURING 6 HOURS)
= 0.9999999999

P (MISSION COMPLETION/BATTLE DAMAGE) = P(3 OF 4 PROC DURING 5 HOURS) x P(2 OF 3 PROC DURING 1 HOUR)
= 0.9999999996 x 0.9999999800
= 0.9999999796

P (SAFE RETURN/NO BATTLE DAMAGE) = P(1 OF 4 PROC DURING 12 HOURS)
= 0.9999999999

P (SAFE RETURN/BATTLE DAMAGE) = P(3 OF 4 PROC DURING 5 HOURS) x P(1 OF 3 PROC DURING 7 HOURS)
= 0.9999999996 x 0.9999999996 = 0.9999999993

FIGURE 15C INTEGRATED BUS ARCHITECTURE - FOUR PROCESSOR SYSTEM



P (MISSION COMPLETION/NO BATTLE DAMAGE) = P(1 OF 2 PROC OF SYS 1 DURING 6 HOURS)
× P(1 OF 2 PROC OF SYS 2 DURING 6 HOURS)
$(0.9999996402)^2 = 0.9999992804$

P (MISSION COMPLETION/BATTLE DAMAGE) = P(1 OF 2 PROC OF SYS 1 DURING 5 HOURS)
× P (1 OF 2 PROC OF SYS 2 DURING 6 HOURS)
× P (1 OF 2 PROC OF SYS 1 DURING 1 HOUR)
× P (1 PROC OF SYS 2 ~ 1 HOUR)
= 0.9990004998 x 0.9999997501 x 0.9999999900 x 0.9999000049
= 0.9989003451

P (SAFE RETURN/NO BATTLE DAMAGE) = P(1 OF 4 PROC DURING 12 HOURS) 0.9999999999

P (SAFE RETURN/BATTLE DAMAGE) = P (1 OF 3 PROC DURING 12 HOURS) = 0.9999999982

FIGURE 16 SPLIT BUS ARCHITECTURE - FOUR PROCESSOR SYSTEM

TABLE 7

EMUX RELIABILITY COMPARISON BUS/PROCESSOR ARCHITECTURE

PROBABILITY FACTOR	INTEGRATED BUS			SPLIT BUS
	TWO PROC.	THREE PROC.	FOUR PROC.	FOUR PROC.
PROBABILITY OF COMPLETING MISSION GIVEN THAT NO BATTLE DAMAGE WILL OCCUR TO PROCESSORS	0.9988007197 (λ_{eff} = 0.02 failures/10^6 hours)	0.9999999982 (λ_{eff} = 300 failures/10^{12} hours)	0.9999999999 (λ_{eff} = 17.0 failures/10^{12} hours)	0.9999992804 (λ_{eff} = 0.12 failures/10^6 hours)
PROBABILITY OF COMPLETING MISSION GIVEN THAT BATTLE DAMAGE WILL OCCUR TO ONE PROCESSOR	0.0	0.9998000189 (λ_{eff} = 33.3 failures/10^6 hours)	0.9999999796 (λ_{eff} = 0.0034 failures/10^6 hours)	0.9989003451 (λ_{eff} = 183.4 failures/10^6 hours)
PROBABILITY OF SAFE RETURN GIVEN THAT NO BATTLE DAMAGE WILL OCCUR TO THE PROCESSORS	0.99999856 (λ_{eff} = 0.12 failures/10^6 hours)	0.9999999982 (λ_{eff} = 150 failures/10^{12} hours)	0.9999999999 (λ_{eff} = 8.3 failures/10^{12} hours)	0.9999999999 (λ_{eff} = 8.3 failures/10^{12} hours)
PROBABILITY OF SAFE RETURN GIVEN THAT BATTLE DAMAGE WILL OCCUR TO ONE PROCESSOR	0.9993 (λ_{eff} = 58 failures/10^6 hours)	0.9999995093 (λ_{eff} = 0.041 failures/10^6 hours)	0.9999999993 (λ_{eff} = 55.8 failures/10^{12} hours)	0.9999999982 (λ_{eff} = 150 failures/10^{12} hours)

TABLE 8

RELIABILITY COMPARISON OF MAIN POWER SOURCE AND PROCESSORS

SYSTEM	RELIABILITY PARAMETERS			MTBF COMPARISON (3 GEN=BASE)
	P (MISSION SUCCESS)	λ_{eff} (Failures/10^6 Hr)	$MTBF_{eff}$ (HOURS)	
FOUR AC GENERATOR SYSTEM	0.9999999365	0.01058	9.45×10^7	63.1
THREE AC GENERATOR SYSTEM	0.999996001	0.6665	1.50×10^6	1.0
THREE PROCESSOR INTEGRATED BUS	0.9999999982	3×10^{-4}	3.3×10^9	2,200
FOUR PROCESSOR INTEGRATED BUS	0.9999999999	17×10^{-6}	5.88×10^{10}	38,200
TWO PROCESSOR/BUS SPLIT BUS	0.9999992804	0.12	8.33×10^6	5.6

The two processor-split bus configuration is marginal by this "order of magnitude" standard. This EMUX arrangement has a factor of five better reliability than the power generation system rather than the desired factor of ten. However, determination of whether the electrical system MTBF is significantly or sufficiently reduced to warrant a third processor in each of the split bus subsystems should be established based on the actual "reliability requirements" for a specific aircraft application. An estimate of the combined reliability of the three generator main power system plus the two processor split bus EMUX arrangement is 0.99999528 for probability of mission completion. This reliability level is sufficient for most applications and is comparable (conservative) to other critical aircraft subsystems.

In summary, for a typical four engine aircraft, either a three processor-integrated bus or a 4 processor-split bus EMUX system arrangement is sufficient in terms of reliability requirements. Some specific aircraft applications may, of course, require significantly higher mission completion requirements, and in turn result in a different system selection. It is also noted that selection of the three processor integrated bus arrangement would also imply the need for three data buses to meet the criteria that one failure not force aborting the mission. Requiring the processor to interface with three data buses results in a slight weight, volume and maintenance failure rate penalty.

The reliability for a single engine aircraft is that of a two processor system shown in Table 7. Note that there is no probability of completing the mission with the loss of one processor since the loss of the remaining processor would not allow safe return of the aircraft.

5.4 ENGINE ELECTRIC START STUDY

A limited trade study was conducted to determine the feasibility of using the power generation system for starting the engine in addition to supplying power to electrical loads. The motor/generator function can be performed with either the cycloconverter VSCF, DC-link VSCF, IDG or CSD technologies. As previously noted, the DC-link and CSD technologies are not considered viable systems for the 1990 time period, consequently were not evaluated for their engine start capabilities in this study.

5.4.1 CYCLOCONVERTER VSCF SYSTEM

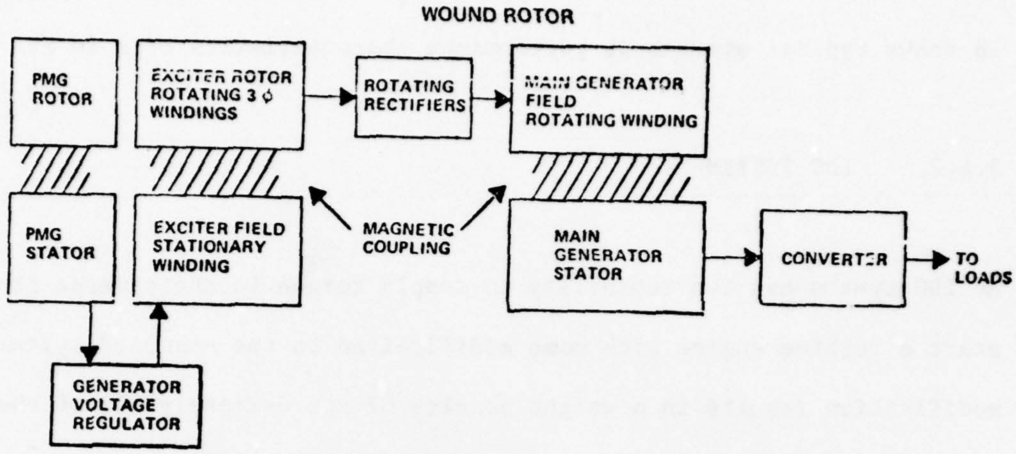
The standard cycloconverter VSCF system can be adapted for the engine start function with the addition of a small amount of control circuits. The circuit modification is primarily in the field excitation area. In the generate mode of operation, the exciter functions as a synchronous generator and provides excitation for the main field. In the motor (start) mode, the exciter is reconnected and the system operates as a wound rotor induction motor. The "dual function" exciter circuit results in a substantial loss of MMF effectiveness and is the prime reason for approximately a 10% increase in system weight.

The system can provide start torque with the machine operating as a synchronous motor or as a brushless DC motor. The performance of a brushless DC motor is similar to a brush type DC motor. The difference between the brush type DC motor and a synchronous motor is that the angle between field flux and armature current is fixed in the DC machine by the geometric relationship of brushes to the field, while this angle varies as a function of power excitation in the synchronous

machine. Maximum power in a synchronous machine occurs when it is on the point of slipping out of synchronism and loss of synchronism results in loss of torque. On the other hand, the DC machine cannot slip out of step because brush position controls the angle. This makes the DC machine practical for operating at optimum torque. A requirement for operating the system equivalent to a brushless DC shunt motor is that the rotor position must be known. This is accomplished with position sensors which monitor rotor position via the PMG field flux.

A simplification of the system is accomplished by using a permanent magnet rotor machine which always supplies its' own excitation. Also, the elimination of rotating rectifiers, rotor cooling components, rotor windings and losses and auxiliary exciters results in a machine with improved reliability and rotor life. The converter package, on the other hand, becomes more complex and results in some added weight. However, the advantages outweigh the disadvantages in applications requiring electric engine start capability. Figure 17 illustrates the comparison in complexity of the two concepts.

A potential application problem of the VSCF starter/generator system is the input power requirement. In the start mode, high currents are required, typically 1/3 higher than in the generate mode. Also the currents are drawn at very low power factors at the beginning of the start cycle and includes large harmonic currents. It is possible that the start power can not be used simultaneously for aircraft loads even with relatively large ground power carts; however, power distortion of the VSCF starter generator is being reduced by improved circuit design. In aircraft equipped with a VSCF APU system having the same rating as the engine driven system, it may be



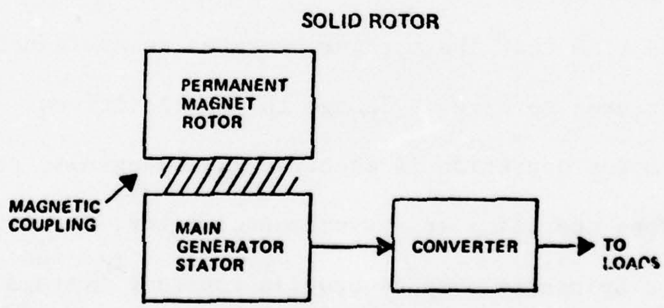


FIGURE 17 WOUND ROTOR VS SOLID ROTOR GENERATOR

necessary to inhibit the undervoltage protection circuit during the start mode since both systems will be operating in their overload ratings. Figure 18 shows typical start mode performance characteristics of a 60 KVA system.

5.4.2 IDG SYSTEM

An IDG system has the capability to supply torque in the reverse direction to start a turbine engine with some modification to the standard system. This modification results in a weight penalty of approximately 10% of the basic system weight, i.e., similar to the VSCF system. Basically, the system operates as follows:

During the start cycle, the generator is brought up to speed as an induction motor. During this time, the reversable displacement pump of the drive is maintained at approximately zero stroke so that very little load is applied to the motor. During the time that the machine operates as an induction motor, the field is short circuted to prevent damage to the rectifiers. When the motor reaches rated speed, motor operation is electrically changed to that of a synchronous motor. When operating as a synchronous motor, the servovalve in the drive controls the hydraulic pump to provide constant maximum working pressure to supply the cranking torque. Cranking torque is maintained past the engine self-sustaining speed until starter cut-out speed is reached to minimize the acceleration time. While operating as a synchronous motor, the voltage regulator supplies an excitation level so that the power into the motor is as near to unity power factor as possible. When the underspeed point of the IDG is reached, the excitation is changed to generator operation.

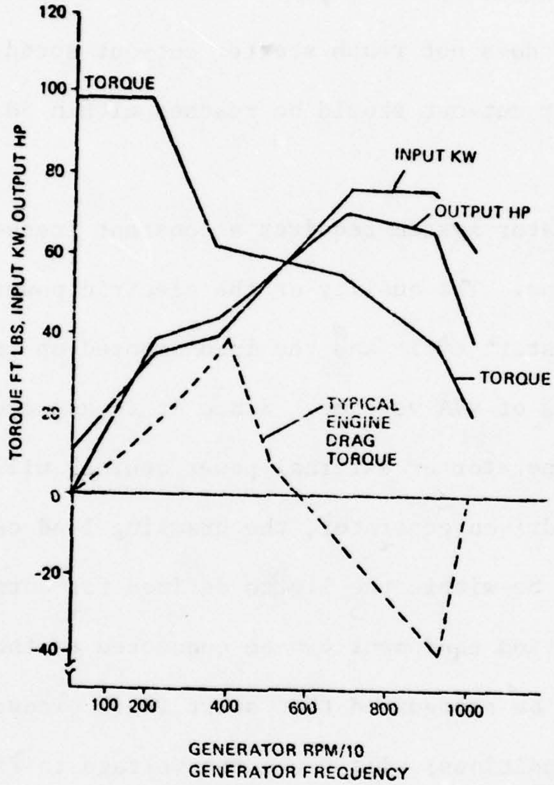


FIGURE 18 TYPICAL PERFORMANCE CHARACTERISTICS OF A 60 KVA VSCF STARTER/GENERATOR SYSTEM.

To provide the starting mode operation, the starter/generator contains two additional protective functions over the standard IDG system.

1. Reactive Current Protection - In the event the starter/generator does not obtain synchronous motor operation 3 to 5 seconds after the initiation of the start, the start is terminated.
2. Start Sequence Protection - If after initiation of a start, the engine does not reach starter cut-out speed; the start is aborted. Starter cut-out should be reached within 30 to 60 seconds.

The IDG starter/generator system requires a constant frequency source of power for starting the engine. The quality of the electric power source is not distorted during the start cycle and the load imposed on the power source can be controlled in terms of KVA vs time. Since it is assumed the power source rating (APU driven generator or external power source) will be equal to the rating on the engine driven generator, the cranking load can be controlled so that the voltage will be within the limits defined for normal operation by MIL-STD-704. Utilization equipment can be connected to the start bus during the start cycle. It must be recognized that start loads exceeding 1.5 per unit rating (fast start conditions) will cause the voltage to fall below MIL-STD-704 normal limits. Figure 19 shows typical performance characteristics of an IDG starter/generator system.

The quantitative analysis which follows was limited to a weight analysis only. In the interest of minimizing the complexity (and cost) of the analysis, a

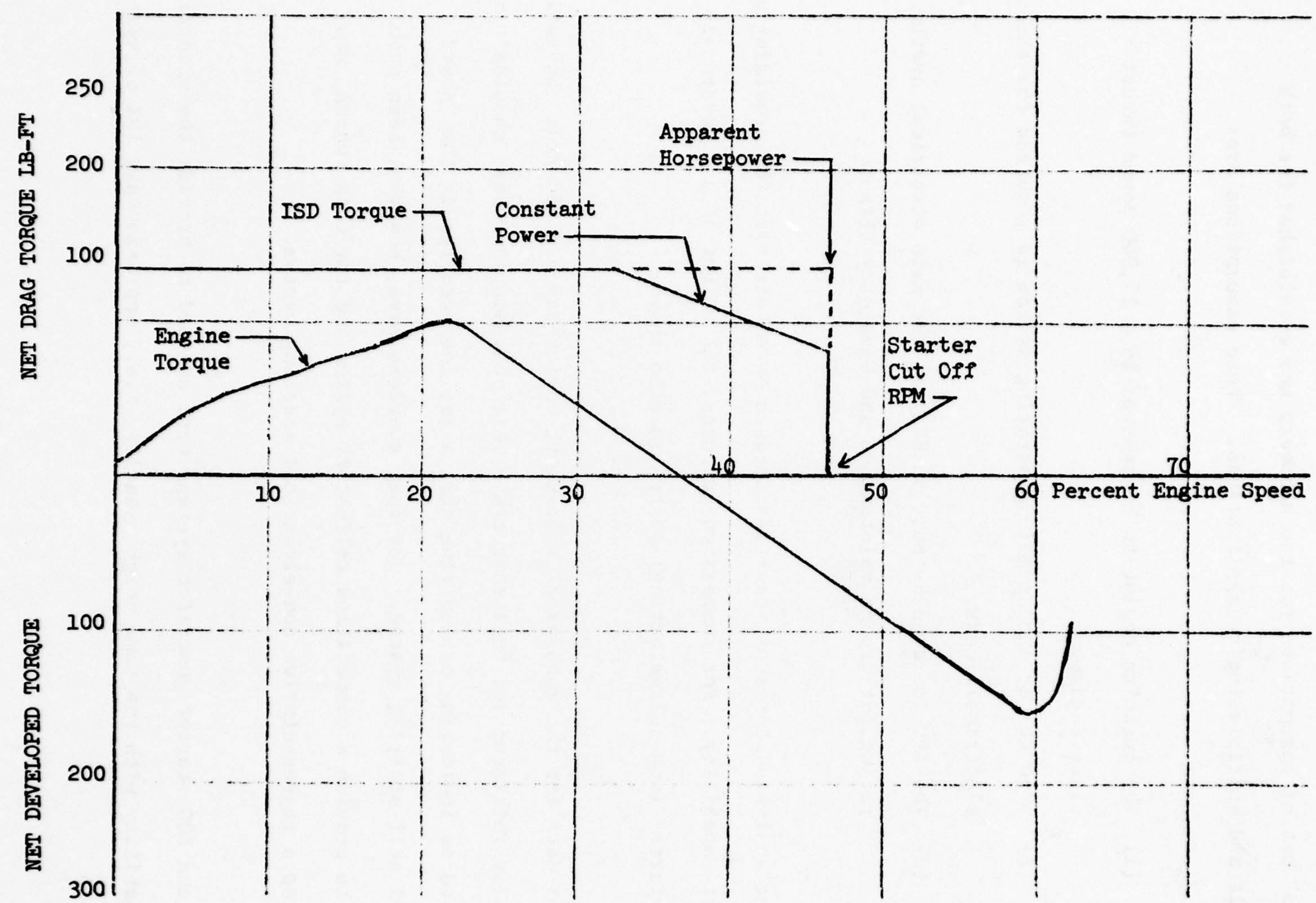


FIGURE 19 TYPICAL PERFORMANCE CHARACTERISTICS OF AN IDG STARTER/GENERATOR SYSTEM.

"standard" set of assumptions for the hardware was estalbished for both the single and multiple engine applications. These assumptions are:

(1) The turbofan engine to be started is a 15,000 pound thrust class engine.

(2) A 60 KVA or larger main generating system is required for the electrical loads.

(3) The APU is rated at 60% (36 KVA) of the main electrical system rating for ground maintenance and emergency flight.

The weight analysis addresses penalties imposed on an aircraft for providing a self-start capability. For comparison purposes, the weight of a standard air turbine starter (non-self-starting) system is also shown.

The weight data for the mechanical self-start option was derived from conventional electrically initiated jet fuel starters. This analysis, however, shouldn't be interpreted as indicating or implying that a jet fuel starter is the "best" mechanical self-starting system. Jet fuel starters have, however, been shown in the past to provide a competitive self-start system and for this reason, was selected as a representative non-electrical starting system.

The VSCF and IDG starter/generator systems were assumed to provide the electric start capability with the same weight penalty, i.e., approximately 10% of system weight.

5.4.1 SINGLE ENGINE AIRCRAFT

Candidate self-starters were first analyzed for the single engine aircraft. The options studied are:

(1) A DC motor initiated jet fuel starter (mechanical start)

(2) A VSCF or IDG starter/generator (electrical start)

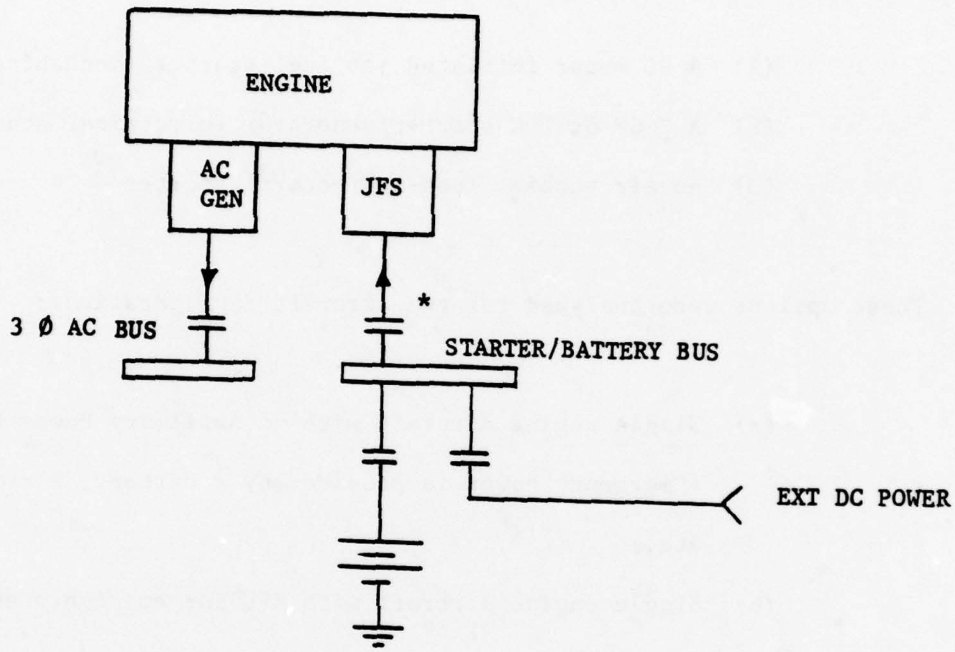
(3) An air turbine (non-self-start) starter

These options were analyzed for two aircraft configurations:

(a) Single engine aircraft with no Auxiliary Power Unit (APU). (Emergency power is provided by a battery, a ram air turbine, etc.)

(b) Single engine aircraft with APU for emergency and ground power.

Figure 20 provides a simplified schematic of the mechanical self-start system for a single engine aircraft. In this configuration, a battery supplies power to a starter bus. A DC contactor switches this bus power to a DC motor contained within the Jet Fuel Starter (JFS). The motor rotates the JFS turbine until JFS ignition occurs. The starter then spins up the engine rotor until a valid start is initiated. The major advantage of the JFS to the electrical system results from the relatively low level of electrical power required to start the engine. In addition, this power can be either AC or DC. The JFS weight and electrical requirements are based on a STU 26/A (military designation) starter. The first



* CONTACTOR ADDED FOR SELF-START CAPABILITY

FIGURE 20 SELF-START FOR AIRCRAFT WITH NO APU DC INITIATED JFS (MECHANICAL)

option column of Table 9 identifies the weight penalties imposed on a single engine aircraft for providing a "mechanical" (JFS) self-start system.

The electrical self-start system illustrated in Figure 21 is comparable to the JFS implemented self-starter. This alternate system utilizes a combination starter/generator to minimize the impact of adding an electric starter. In the past, this starter/generator combination dictated a DC machine. This occurred because a practical sized AC motor lacks sufficient stall torque (typically, several hundred foot-pounds) to start engine rotor rotation. Since approximately 85 percent of the power on conventional aircraft is distributed as AC power, a large static inverter is required if a DC generator/starter is used.

With the development of brushless DC motor concepts, particularly as applied to VSCF systems, a combination of "DC motor" and "AC generator" functions can be provided by one machine. Thus, use of an electric starter no longer dictates a DC power generator.

The starter system illustrated in Figure 21 utilizes a VSCF starter/generator which creates AC power via an electronic converter in the generate mode. In the start mode, AC power is chopped and controlled to the various generator windings to effect the equivalent of a brushless DC motor. In the case of an IDG system, the machine is operated as an induction/synchronous motor. The major disadvantage of both approaches results from requiring AC power of a significant quantity to start the engine. For a true self-start capability, this AC power must be derived from a battery or from an APU. Table 9 addresses the weight penalties

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TABLE 9

ENGINE SELF START WEIGHT COMPARISON (NO APU)

(SINGLE 15,000 LB. THRUST ENGINE ~NO APU)

ITEM	OPTION (WT IN LBS)		
	SELF-START SYSTEM		NO SELF-START
	MECHANICAL (JFS)	ELECTRICAL	AIR TURBINE
GENERATOR Δ	0	13	0
BATTERY Δ	33	282	0
STARTER - JFS	92	0	0
AIR TURBINE STARTER (M19557/5)	0	0	23
INVERTER or DRIVE Δ	0	100	0
CONTACTOR Δ	2	4	0
TOTAL Δ	127	399	23 (REF ONLY)
SAVINGS	(68%) 272 LBS		

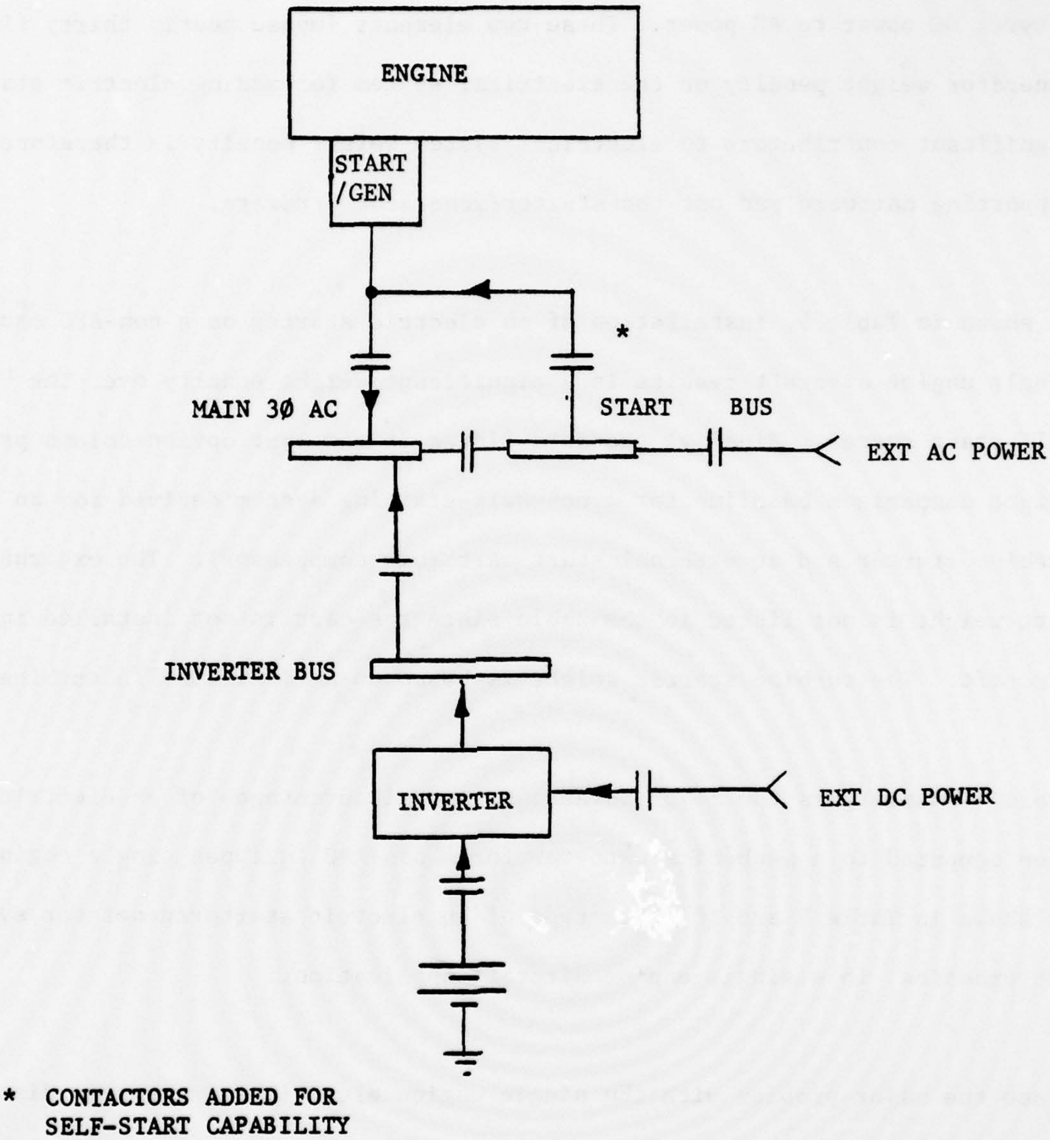


FIGURE 21 SELF-START FOR AIRCRAFT WITH NO APU STARTER/GENERATOR (ELECTRICAL)

resulting from a battery powered electric start system while Table 10 addresses APU sourced starting. The Table 9 data reflects the significant weight penalty imposed not only by the battery but also by the static inverter required to convert DC power to AC power. These two elements impose nearly thirty times the generator weight penalty on the electrical system for adding electric start. The significant contributors to electrical system weight penalty is therefore the supporting hardware and not the starter/generator hardware.

As shown in Table 9, installation of an electric starter on a non-APU equipped single engine aircraft results in a significant weight penalty over the "mechanical" self-start system. Finally, the data listed in the last option column provides a weight comparison baseline for a non-self-starting system derived for an air turbine starter and an external start cart (air compressor). The external start-cart weight is not listed in the table since the cart is not installed in the aircraft. The turbine starter weight is based on a MIL-T-19557/5 turbine.

Table 10 summarizes the major advantages and disadvantages of an electric starter when compared to a mechanical starter for a non-APU equipped single engine aircraft. As shown in Table 9 and 10, this type of an electric starter/generator system is not practical in a single engine aircraft application.

Since the major problem with the single engine electric start system discussed above resulted from starting power source penalties, a second single engine arrangement was investigated. In this second arrangement, an APU is installed in the aircraft for emergency and ground maintenance requirements. Figures 22 and 23 depict typical schematics for the mechanical and electrical self-start

TABLE 10

ADVANTAGES/DISADVANTAGES OF ELECTRIC SELF-START
VS
MECHANICAL (JFS) SELF-START
FOR
A NON-APU EQUIPPED SINGLE ENGINE AIRCRAFT

ADVANTAGE	DISADVANTAGE
1. Reduce space(frontal area) required at the engine accessory drive.	1. Larger capacity generator required to provide high motoring torque.
2. Reduces maintenance failure rates by eliminating separate starting hardware.	2. Requires installation and maintenance of large batteries and inverters.
3. Improves overall weapon system reliability by reducing parts count.	3. May distort aircraft power quality during start mode to a more severe level than the JFS system.
4. Provides faster starting since the two "gas turbines" of the JFS configuration are not being started sequentially (i.e., JFS turbine starts then engine starts.)	4. Redesign of external ac power cart likely required if external ac power starts are attempted.
5. No additional connections to the engine required other than for the generator.	5. Significantly increases the weight of the electrical system.
6. Not as susceptible to altitude limitations for inflight restart.	6. Cannot start engine in the event of a system failure. An alternate means to start the engine in addition to the start/gen may be desired.

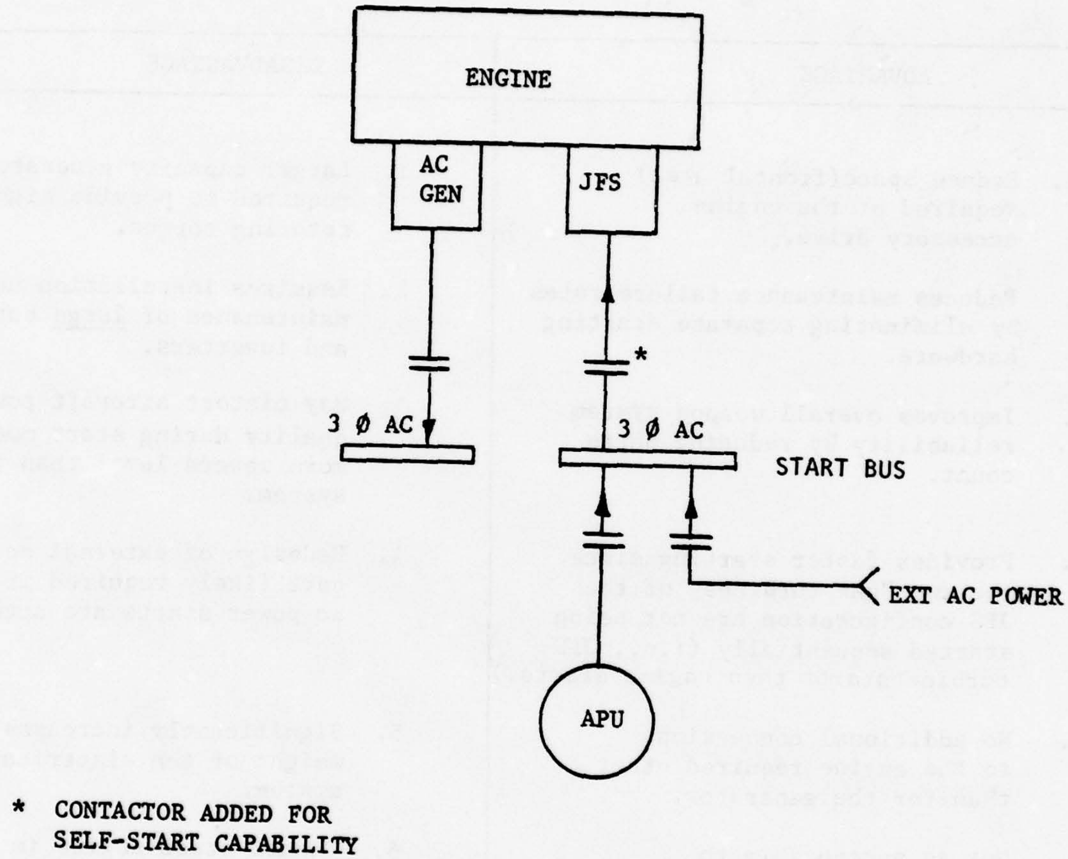
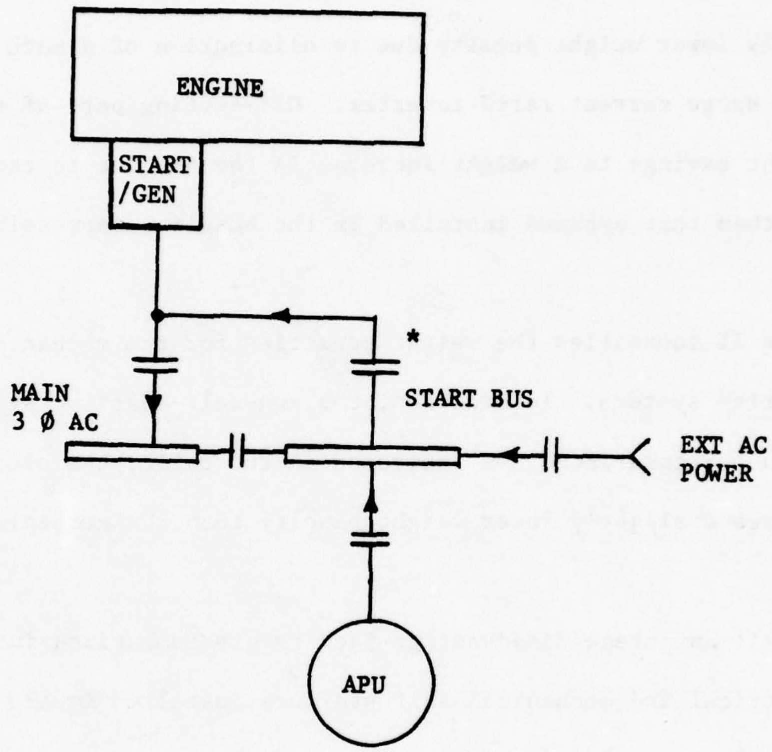


FIGURE 22 SELF-START FOR APU EQUIPPED AIRCRAFT AC INITIATED JFS (MECHANICAL)



* CONTACTOR ADDED FOR SELF-START CAPABILITY

FIGURE 23 SELF-START FOR APU EQUIPPED AIRCRAFT STARTER/GENERATOR (ELECTRICAL)

systems respectively in an APU equipped single engine aircraft. The only difference between these two systems and those defined in Figures 20 and 21 is the elimination of the battery and iverter power source by the APU. The JFS implemented system has a 33 percent lower weight penalty due to battery elimination. The electric start implemented system, however, has a significantly lower weight penalty due to elimination of a much larger battery plus a high surge current rated inverter. Off-setting part of the battery/inverter weight savings is a weight increase in the APU due to requiring a higher rated APU than that assumed installed in the baseline (non-self-start) aircraft.

Table 11 identifies the weight penalties for the mechanical and electrical self-starting systems. In addition, the non-self-starting system weight is also shown for comarison. As indicated in the table, the electrical starting system imposes a slightly lower weight panelty than the mechanical system.

Overall advantage/disadvantage factors are summarized in Table 12 for the electrical and mechanical self-starters installed in APU equipped single engine aircraft.

In general, the electric starter on an APU equipped, single engine aircraft is marginally "better" than a JFS implemented system. The electric starter is evaluated as marginally better because another mechanical start option is available. However, it does not meet the established requirement of being independent of the engine. This option consists of an APU mounted directly on the engine accessory drive. The APU design is such that engine starting is provided directly by a mechanical link from the APU. This alternate was not

TABLE 11

ENGINE SELF-START WEIGHT COMPARISON

(SINGLE 15,000 LB. THRUST ENGINE ~ WITH APU)

ITEM	OPTION (WT IN LBS)		
	SELF-START SYSTEM		NON SELF-START
	MECHANICAL (JFS)	ELECTRICAL	AIR TURBINE
GENERATOR Δ	0	13	0
STARTER (JFS)	92	0	0
AIR TURBINE STARTER	0	0	23
CONTACTOR Δ	1	1	0
APU Δ	0	48	0
TOTAL Δ	93	62	23 (REF ONLY)
SAVINGS		(33%) 31 LBS	

TABLE 12

ADVANTAGES/DISADVANTAGES OF ELECTRICAL VERSUS MECHANICAL SELF-STARTING FOR AN APU EQUIPPED SINGLE ENGINE AIRCRAFT

ADVANTAGE	DISADVANTAGE
1. REDUCES SPACE (FRONTAL AREA) REQUIRED AT THE ENGINE ACCESSORY DRIVE.	1. LARGER GENERATOR CAPACITY REQUIRED TO PROVIDE FOR HIGH MOTORING TORQUES.
2. REDUCES MAINTENANCE FAILURE RATES BY ELIMINATING DEDICATED STARTING HARDWARE.	2. LARGER APU CAPACITY REQUIRED TO PROVIDE STARTING POWER
3. IMPROVES OVERALL WEAPON SYSTEM RELIABILITY BY REDUCING PARTS COUNT.	3. MAY DISTORT ELECTRICAL POWER QUALITY DURING START MODE TO A MORE SEVERE LEVEL THAN WITH A JFS STARTER.
4. PROVIDES FASTER STARTING THAN JFS.	4. REDESIGN OF EXTERNAL AC POWER CART REQUIRED.
5. PROVIDES LOWER WEIGHT PENALTY FOR SELF-STARTING	5. OPTION IS NOT AVAILABLE FOR AN EMERGENCY GROUND START WITH COMPRESSED AIR IN THE EVENT OF SYSTEM FAILURE. AN ALTERNATE MEANS FOR STARTING THE ENGINE MAY BE DESIRABLE IN SOME APPLICATIONS.
6. NO ADDITIONAL CONNECTIONS REQUIRED TO ENGINE AREA.	

analyzed in detail, but a weight estimate was made with the assumption that the "initially installed" APU capacity is sufficient to start the engine. This option yields a projected weight penalty of 1 pound versus the 62 pound penalty of an electric starter. This lesser weight penalty not only assumes that the initial APU capacity is sufficient for engine starting, but that the mechanical link to the engine does not impose additional penalties. This second assumption is a function of installation restrictions. In addition, this option assumes that the reduced safe return probability resulting from higher system vulnerability and susceptibility due to single failures is acceptable. Higher vulnerability results from less physical isolation between primary and emergency power sources.

5.4.2 FOUR ENGINE AIRCRAFT

Similar trades between mechanical and electrical self-starting systems can be conducted for four engine aircraft. The major impact in transiting from single to multiple (four) engine aircraft is in spreading the "cost" of the self-starting power source over more engines (or more power channels).

Figures 24 and 25 schematically illustrate mechanical (JFS) and electrical self-start systems, respectively, for a four engine aircraft which is not equipped with an APU. These two configurations are comparable to the single engine self-start systems of Figures 20 and 21.

Table 13 identifies the principle weight penalty contributors for both mechanical and electrical systems. As shown in the table, the battery and inverter penalty

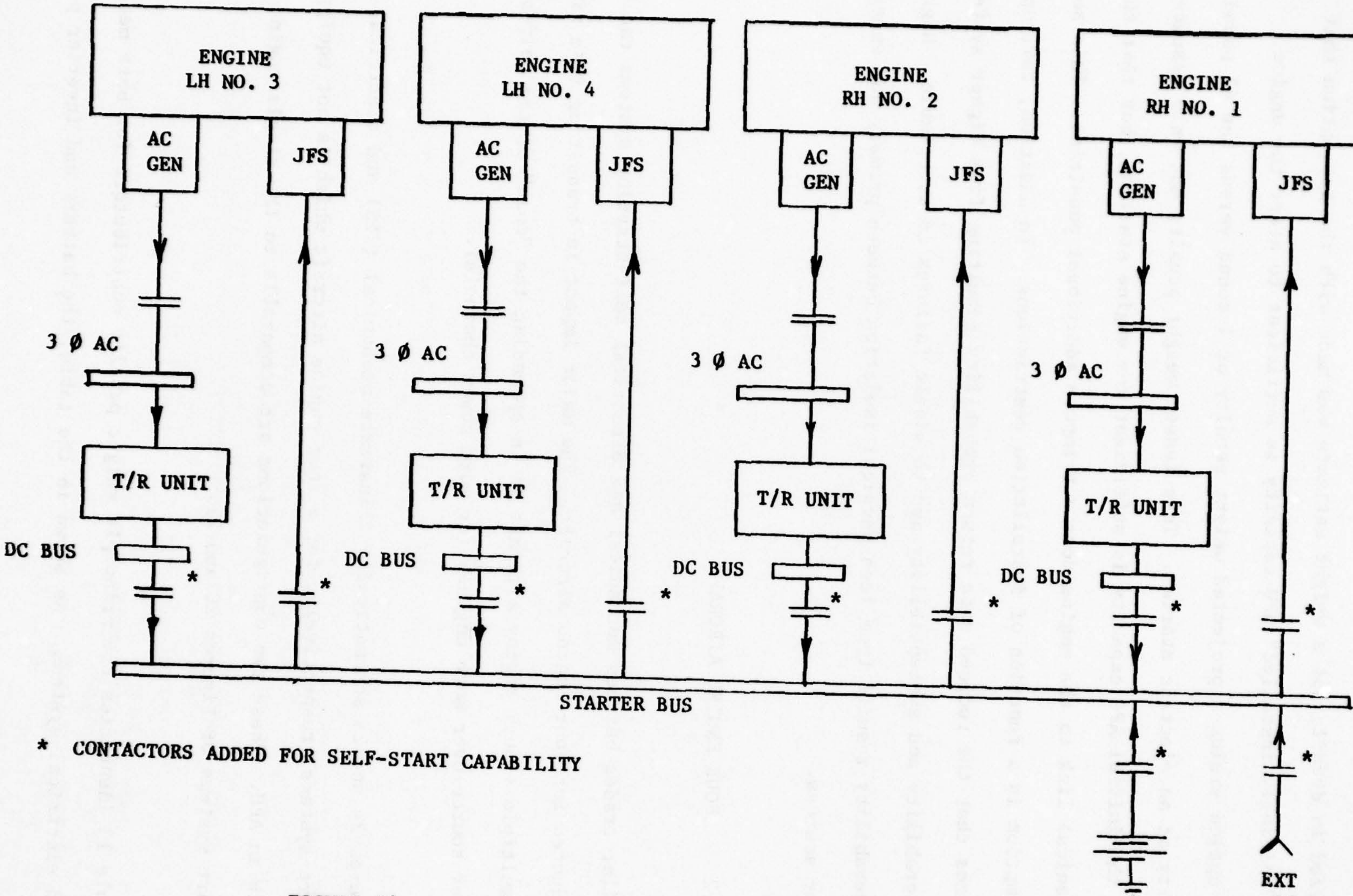


FIGURE 24 SELF-START FOR AIRCRAFT WITH NO APU - DC INITIATED JFS (MECHANICAL)

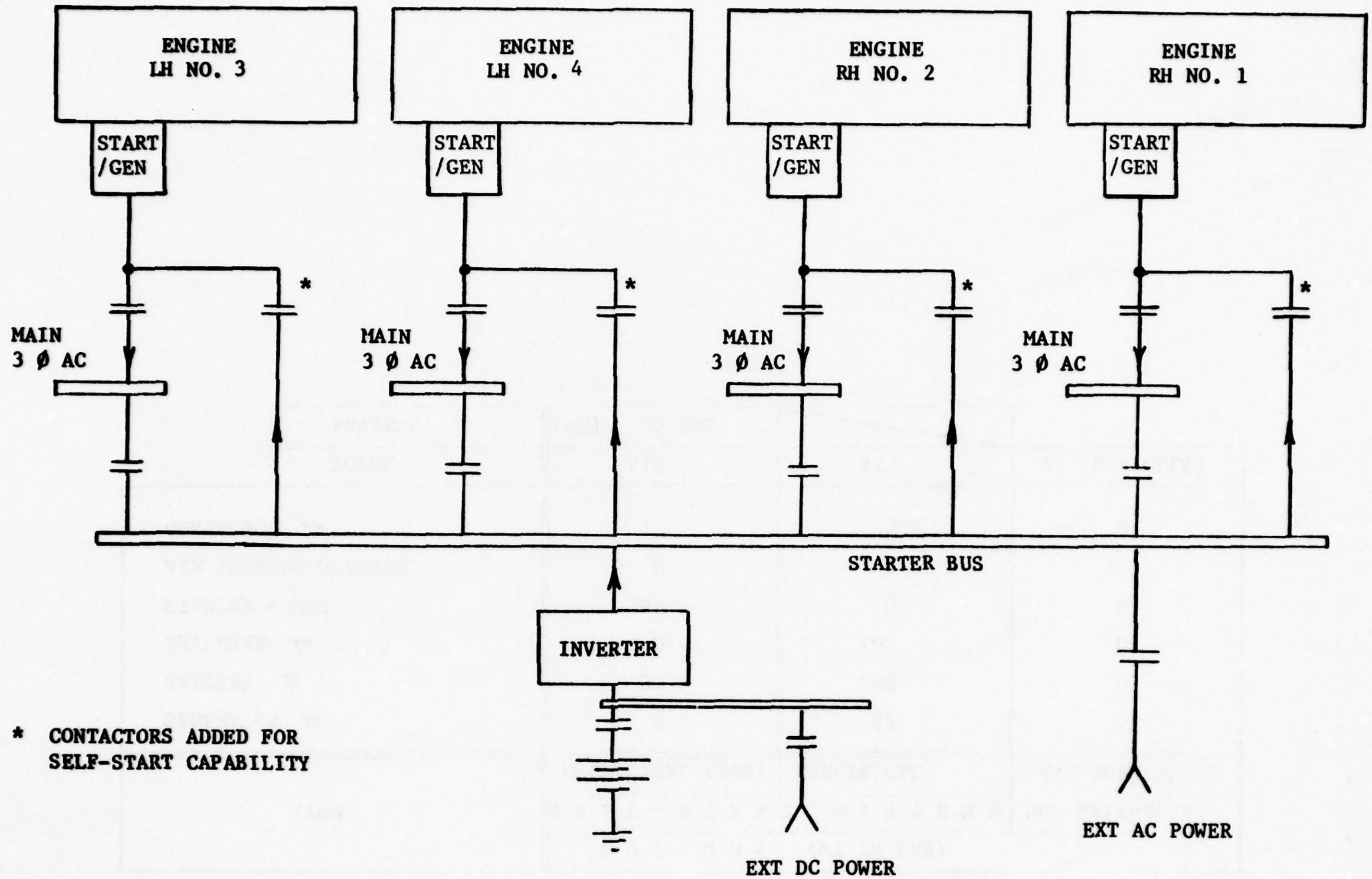


FIGURE 25 SELF-START FOR AIRCRAFT WITH NO APU - STARTER/GENERATOR (ELECTRICAL)

TABLE 13

WEIGHT COMPARISON

(FOUR 15,000 LB. THRUST ENGINES ~ NO APU)

ITEM	OPTIONS (WT IN LBS)		
	SELF-START SYSTEMS		NON SELF-START
	MECHANICAL (JFS)	ELECTRICAL	AIR TURBINE
GENERATOR △	0	52	0
BATTERY △	33	282	0
INVERTER △	0	100	0
STARTER - JFS	368	0	0
AIR TURBINE STARTER	0	0	92
CONTACTOR △	17	8.5	0
TOTAL	418	443	92 (REF ONLY)
SAVINGS	(6%) 25 LBS		

imposed on the electric start system again indicates a slight advantage in installing the mechanical system as was determined in the single engine application. A summary of advantages and disadvantages for installing an electric starter as opposed to a "mechanical" starter in a four engine (non-APU) aircraft is presented in Table 14.

Figures 26 and 27 illustrate comparable mechanical and electrical self-start systems for APU equipped four engine aircraft. These two systems are equivalent to the single engine configurations of Figures 22 and 23.

As shown in Table 15, a significant weight advantage results from selecting the electric start system for the APU equipped four engine aircraft. When coupled with the advantages listed in Table 16, an electric starter appears as the desirable choice for multi-engine self-starting when an APU is installed. In fact, the weight saved by selecting the electric start system could, in some cases, cancel the entire weight of the installed APU.

In summary, a qualitative comparison of electrical versus mechanical self-starting systems for the four aircraft configurations is shown in Table 17. Although the advantage of electric starters is marginal on some single and multi-engine aircraft applications, an electric starting system should be included in the baseline design for study of its' stability impact on the electric power system.

TABLE 14

ADVANTAGES/DISADVANTAGES FOR ELECTRIC START OVER MECHANICAL START IN A NON-APU-EQUIPPED FOUR ENGINE AIRCRAFT

ADVANTAGES	DISADVANTAGES
1. Reduces space (frontal area) required at engine accessory drive.	1. Larger generator capacity required at each engine to provide the high motoring torques.
2. Reduces maintenance failure rate by eliminating dedicated starting hardware on each engine.	2. Installation and maintenance of large battery and inverter required.
3. Improves overall weapon system reliability due to reduced parts count.	3. Greater electrical power quality distortion during start mode.
4. Provides faster engine start.	4. May require external ac power cart redesign.
5. No additional electrical or fuel line connections required in vicinity of engines.	5. Requires installation of starter/generator on each engine. (Mission completion requirements may only require three generators)
6. Less susceptible to high altitude starting limitations.	6. Emergency ground start with compressed air is not practical if starter fails. An alternate means for starting the engine may be desirable for some applications.
7. Aids in justifying fourth generator for improved mission completion/safe return probabilities.	7. Slight weight penalty (10%).
8. Greater growth capability as result of larger capacity generating system.	

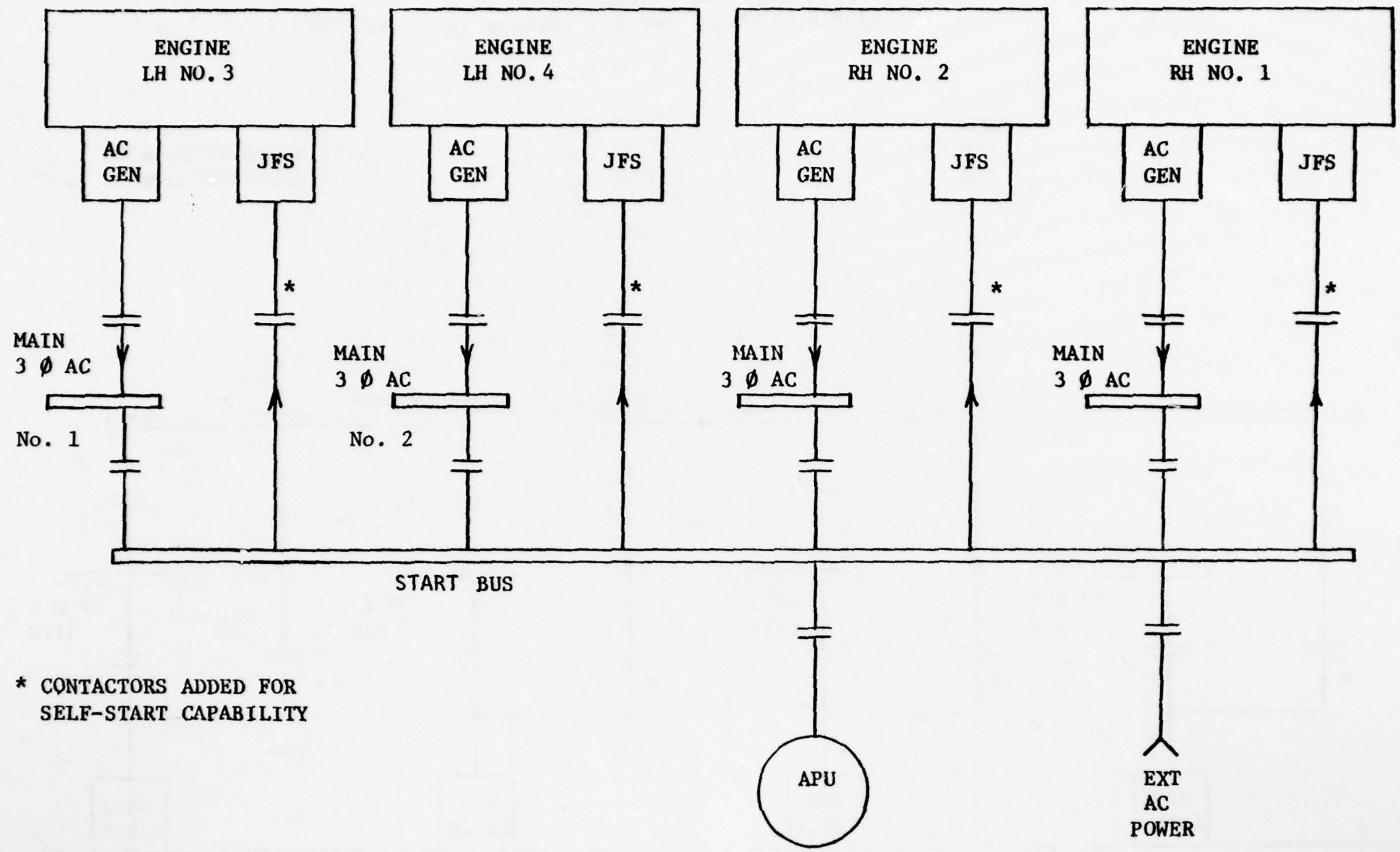


FIGURE 26 SELF-START FOR APU EQUIPPED AIRCRAFT - AC INITIATED JFS (MECHANICAL)

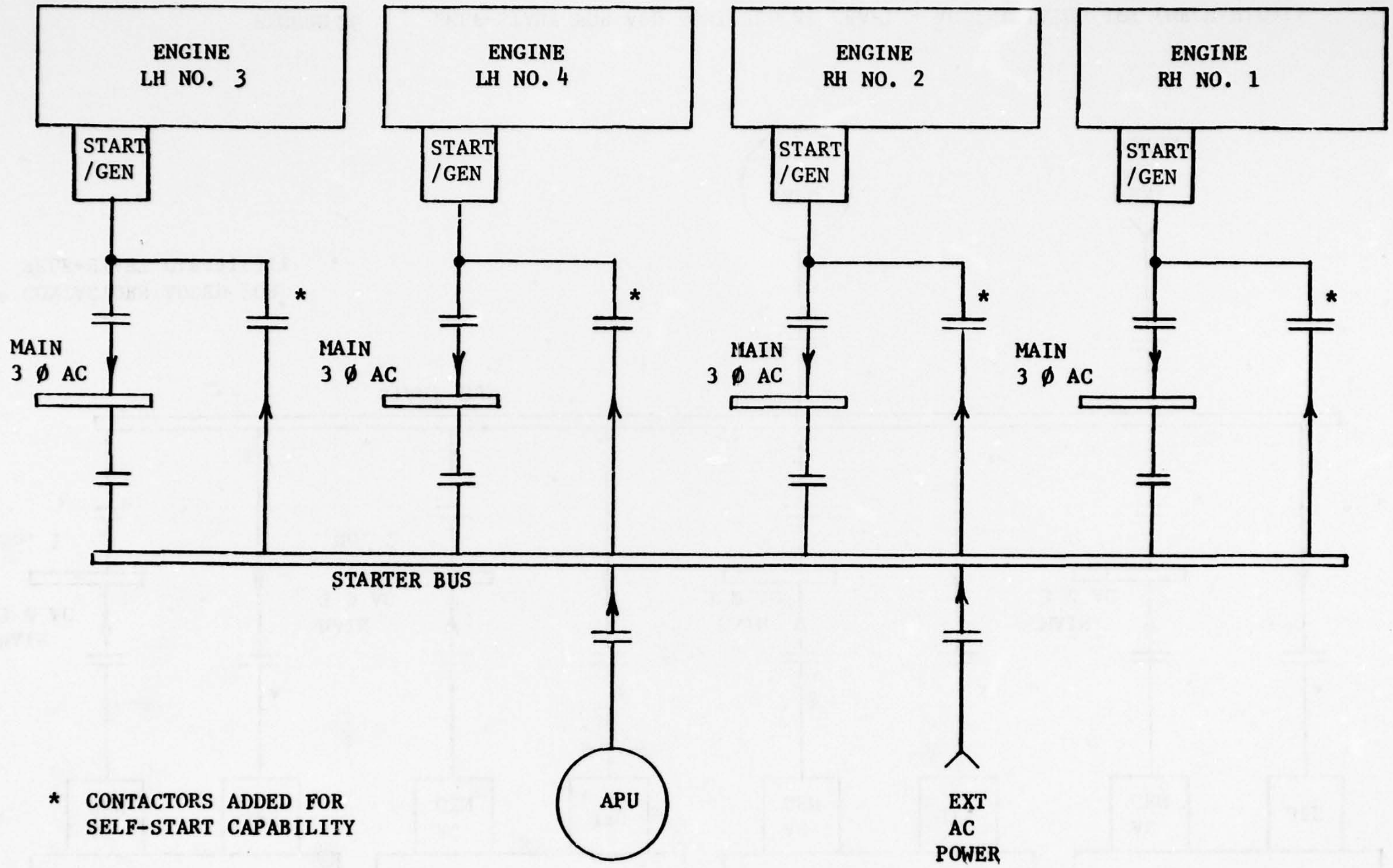


FIGURE 27 SELF-START FOR APU EQUIPPED AIRCRAFT STARTER/GENERATOR (ELECTRICAL)

TABLE 15

WEIGHT COMPARISON

(FOUR 15,000 LB. THRUST ENGINES - WITH APU)

ITEM	OPTION (WT IN LBS)		
	SELF-START SYSTEMS		NON SELF-START
	MECHANICAL (JFS)	ELECTRICAL	AIR TURBINE
GENERATOR △	0	52	0
APU △	0	48	0
STARTER - JFS	368	0	0
AIR TURBINE STARTER	0	0	92
CONTACTOR △	3	3	
TOTAL	371	103	92 (REF ONLY)
SAVINGS		(72%) 268 LBS	

TABLE 16

ADVANTAGES/DISADVANTAGES OF ELECTRIC START VERSUS MECHANICAL START IN APU EQUIPPED FOUR ENGINE AIRCRAFT

ADVANTAGES	DISADVANTAGES
1. Reduces space (frontal area) required at engine accessory drive.	1. Larger generator capacity required at each engine to provide high motoring torques.
2. Reduces maintenance failure rates by eliminating dedicated starting hardware on each engine.	2. Larger APU capacity required to provide electric starting power.
3. Improves overall weapon system reliability due to reduced parts count.	3. Greater electrical power quality distortion during starting mode.
4. Provides faster engine start than JFS system.	4. Redesign of external ac power cart likely required.
5. No additional electrical or fuel line connections required at the engines.	5. Requires installation of starter/generator on each engine (Mission requirements may only require three generators)
6. Less susceptible to high altitude starting limitations.	6. Emergency ground start with compressed air is not practical if starter fails. Alternate means for starting the engine may be desired for some applications.
7. Aids in justifying fourth generator for improved mission completion/safe return probabilities.	
8. Significant weight savings.	
9. Greater growth capacity as result of larger capacity generating system.	

TABLE 17

SELF-START QUALITATIVE COMPARISON MATRIX

AIRCRAFT CONFIGURATION	MECHANICAL	ELECTRICAL
Single Engine No APU	Significant Advantages	Minimal Advantages
Single Engine With APU	Marginally Advantageous under certain conditions.	Marginally advantageous under certain conditions
Four Engine No APU	Marginally Advantageous under certain conditions	Marginally advantageous under certain conditions.
Four Engine With APU	Minimal Advantages	Significant Advantages

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5.5 EMUX INTEGRATED GENERATOR CONTROL CONCEPTS

EMUX integrated generator control concepts were evaluated for the multi-engine aircraft. For this study, it is assumed the electrical system consists of three main generators, one APU driven emergency generator and two battery/inverter power sources. However, the concepts are applicable to any number of generator systems with minimal change.

The four control concepts evaluated are summarized by the following characteristics:

(1) Conventional dedicated Generator Control Units (GCU), Bus Control Units (BCU), and EMUX power distribution.

(2) EMUX central processor control of GCU and BCU functions in addition to power distribution functions.

(3) GCU and BCU processing at remote EMUX "smart" terminals with supplementary processing at EMUX central processors.

(4) GCU and BCU processing provided by dedicated G/BCU smart terminals with MIL-STD-1553 interface to EMUX.

Figure 28 shows a simple control implementation (concept 1) using conventional generator control architectures. This configuration consists of dedicated Generator Control Units (GCU's), Bus Control Units (BCU's), and EMUX hardware. The system arrangement permits real time GCU control of the generator regulation and protective functions as well as real time interface between GCU channels for system paralleling (load division) and synchronizing functions. Also, the dedicated BCU hardware provides real time bus management and bus/feeder protection.

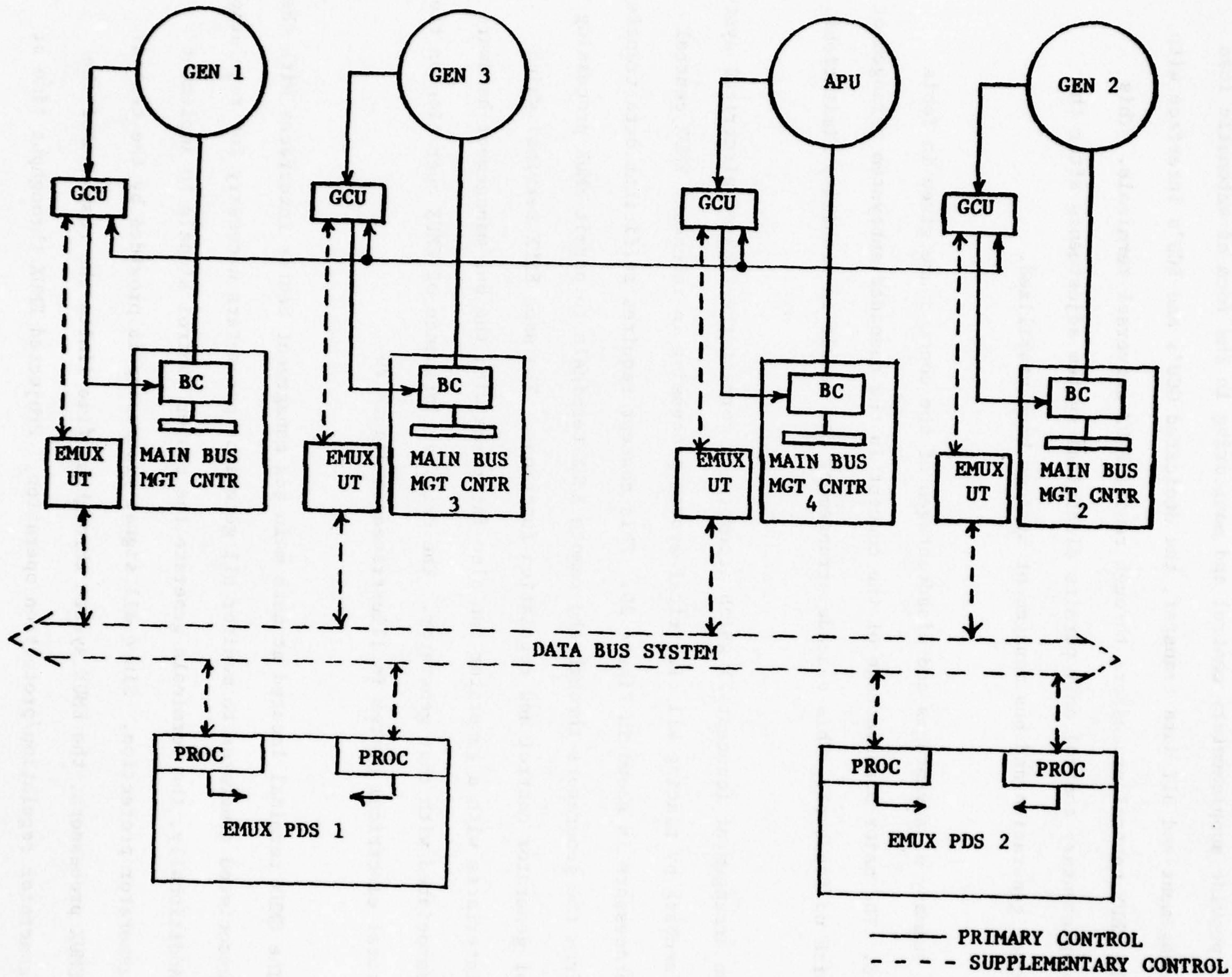

FIGURE 28 EMUX/GENERATOR CONTROL - CONCEPT 1

To provide supplementary control and monitoring in the form of automatic load management and BIT data transfer, the dedicated GCU's and BCU's interface with the EMUX central processors through remote EMUX universal terminals. This supplementary control only permits electrical system adjustments after the power generation and bus management systems have stabilized.

A summary of advantages and disadvantages of the concept are given in Table 18. The major disadvantage of the concept is the cascaded subsystem arrangement with no feedback. This cascade arrangement may result in stability limitations.

An arrangement (concept 2) which attempts to correct the lack of electrical system feedback by placing all electrical system processing in redundant EMUX central processors is shown in Figure 29. This concept requires sufficient data transfer from the generators through the remote EMUX terminals to permit EMUX processing of generator control and regulation functions. The same EMUX terminal which interfaces with a generator can also interface with the bus management hardware associated with that generator. The critical influence of EMUX operation on the total electrical system is illustrated in Figure 29.

The EMUX terminal located at each main bus management center interfaces with the associated generator to monitor all generator parameters necessary for regulation. Additionally, the terminals generate the proper control signals to implement generator protection. Since all signal processing is provided by the central EMUX processors, the EMUX system throughput time limits the response of the generator regulation/protection operation. Projected EMUX throughput time of 20 to 50 milliseconds is several orders of magnitude too slow for proper generator control necessary for high quality power and paralleling capability.

Table 18

EMUX/GENERATOR CONTROL - CONCEPT 1

ADVANTAGES	DISADVANTAGES
1. Dedicated hardware can be optimized for dedicated function.	1. Requires use of EMUX universal terminals to interface GCU and BCU functions with the EMUX processors.
2. Interfaces simplified by minimizing data transfer between subsystems.	2. Cascaded system is essentially open loop with minimal feedback thru EMUX data bus. This decreases overall system stabilities.
3. Minimum redesign of existing hardware required.	3. Response of data transfer loop thru the EMUX processor is too slow to adequately control any action except load management (EMUX thruput time is estimated at 50 milliseconds as opposed to ~50 microseconds generator data transfer requirements.).
4. Since operation is an isolated subsystem, susceptability to failures which completely shutdown the electrical system is very low.	4. Separate dedicated units may result in higher system weight, failure rate and maintenance actions.
5. Direct interface between GCU's permit rapid parallelling and synchronizing response.	
6. GCU and BCU functions can be implemented with μ processors or discrete/MSI/LSI circuitry.	
7. Critical loads can be powered (uncontrolled) even without EMUX operating by using "normally on" load controllers.	

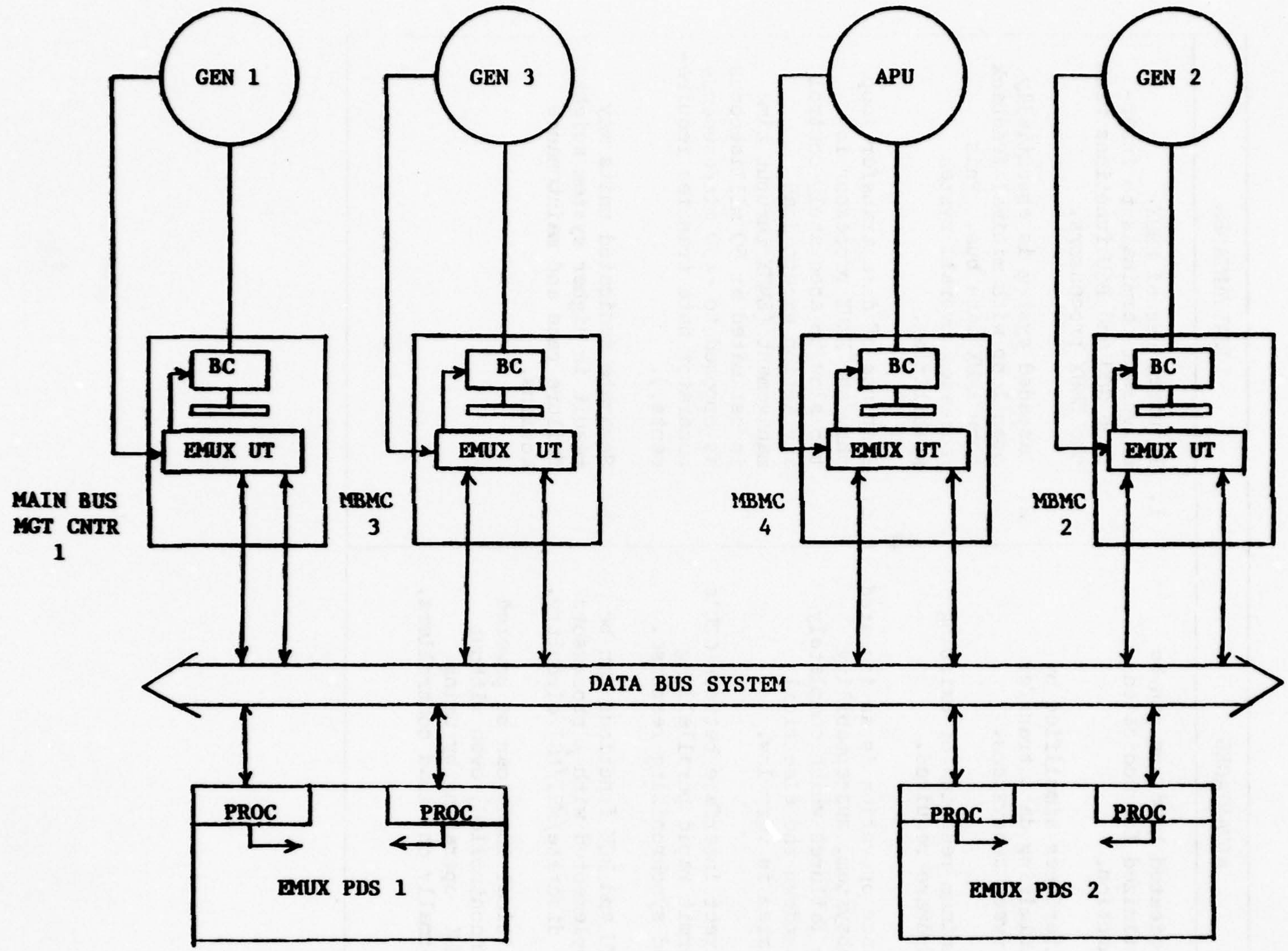

FIGURE 29 EMUX/GENERATOR CONTROL - CONCEPT 2

This slow response of EMUX is the major disadvantage of using EMUX in a real-time generator control loop. A summary of advantages and disadvantages of this concept is presented in Table 19.

An arrangement (concept 3) which alleviates the response time problem is shown in Figure 30. This concept basically is the same as that shown in Figure 29 except that generator/bus control signal processing is performed by remote EMUX "smart" terminals rather than by the central processors. By performing generator control signal processing at local terminals, the data transfer requirements over EMUX buses are minimized. This, in turn, speeds up the system data throughput time. However, the EMUX discrete I/O interface must still be adapted to the generator analog signals. The higher information content of the generator interface I/O data still cannot be effectively handled by a discrete input formatted EMUX remote terminal.

This concept yields a lower vulnerability over the concept shown in Figure 29 since only the remote terminal associated with each generator must be powered by the respective generator PMG. Note that the concept shown in Figure 29 requires the PMG to supply power to the EMUX processors in addition to the remote terminals. The major advantages and disadvantages of the concept shown in Figure 30 are given in Table 20. The major problems to be confronted by this third concept is the attempt to implement generator control through a "standard" EMUX I/O interface and to provide the interface between generators over the "slow" throughput EMUX system. To overcome these problems requires the use of "non-standard" smart data terminals at the main bus centers as shown in Figure 31. The predominant characteristics of this concept is the use of modular smart data

Table 19

EMUX/GENERATOR CONTROL - CONCEPT 2

ADVANTAGES	DISADVANTAGES
1. Complete use of standard hardware for data transfer and I/O interfaces. 2. All power control processing would be accomplished at central locations, this would increase overall flexibility and subsystem coordination.	1. A/D conversion required at generator and current sensor for data transfer to EMUX. It is not likely that these interfaces would convert to digital if dedicated GCU's were used. Excessive channel requirements may result from A/D conversion. 2. Thruput time for EMUX (as presently defined) is insufficient to control/regulate generators in "real time". 3. Generator PMG's or alternate continuous power sources are required to power up EMUX prior to regulation and control of the generators. It is estimated that approximately 500 watts (of PMG power) will be required to power-up sufficient EMUX hardware to operate generators. 4. Not possible to power any loads if EMUX is not operational. 5. Requires addition of EMUX universal terminals in each main bus center for generator interface.

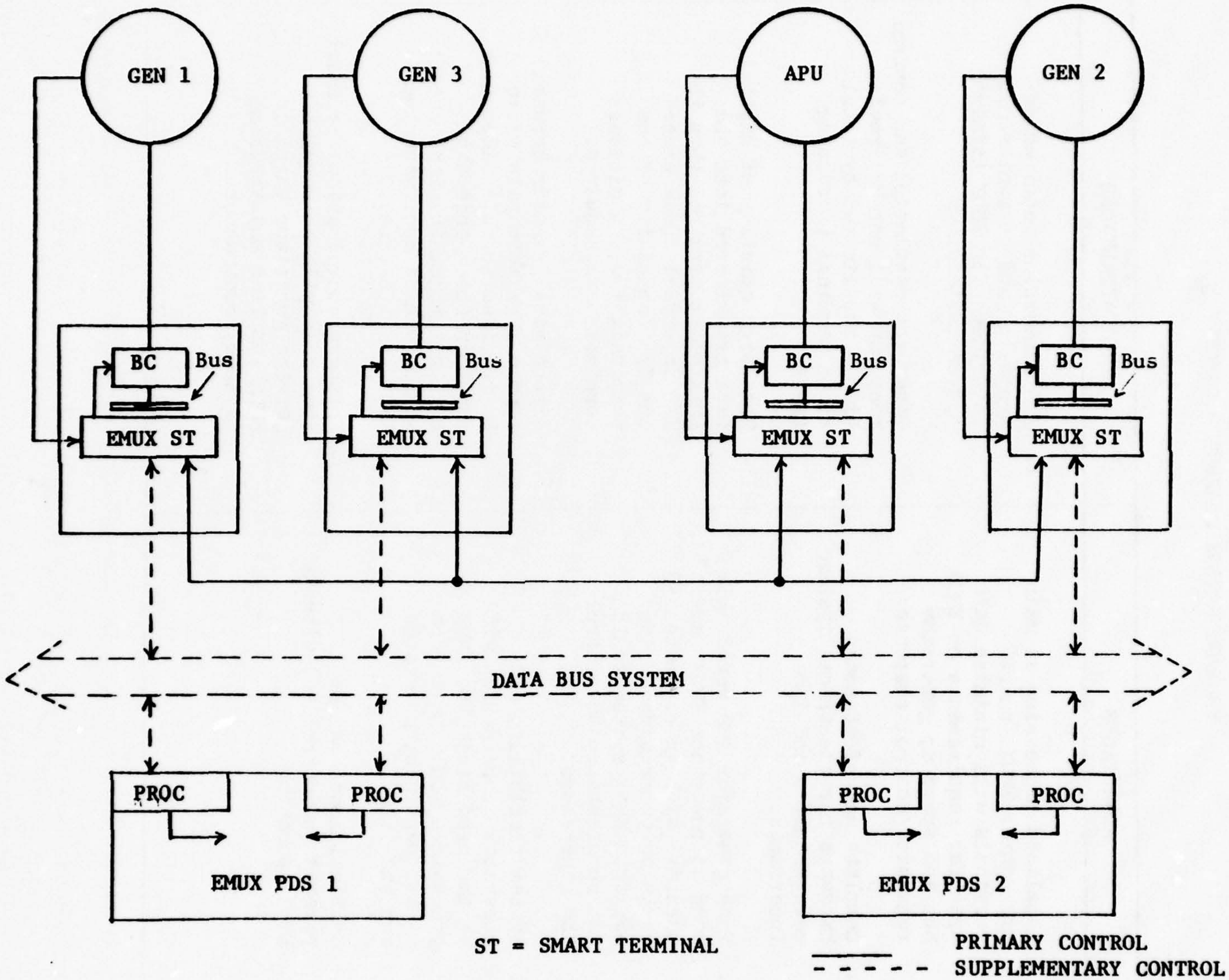

FIGURE 30 EMUX/GENERATOR CONTROL - CONCEPT 3

Table 20

EMUX/GENERATOR CONTROL - CONCEPT 3

ADVANTAGES	DISADVANTAGES
1. Localized processing at main bus center EMUX "smart" terminals will minimize data transfer requirements on EMUX bus and speed up generator regulator/control response. 2. Complete use of standard hardware for electrical system data transfer and I/O interfaces. 3. Each generator PMG would only need to power one EMUX "smart" terminal with an estimated 30 watts power required. The complete EMUX system would not be necessary to control the generators. 4. System flexibility and subsystem coordination will be improved since the EMUX processors would have access to all pertinent generator and bus data. 5. Critical loads can be powered (uncontrolled) without EMUX operating.	1. A/D conversion required at generator and current sensor interfaces for EMUX terminal I/O compatibility. 2. Generator regulation and control response will not be "real time", but limited by total smart terminal processing tasks. 3. The large quantity of digital data transferred into the EMUX smart terminal thru the I/O interface could exceed the I/O capability of one terminal, i.e., additional terminals are required. 4. Direct data transfer between remote EMUX terminals would be required to fully implement this concept. This data terminal communication concept would require modifying the EMUX architecture. 5. Software coordination for smart terminals will be required between generator supplier, EMUX supplier and airframe (system) contractor.

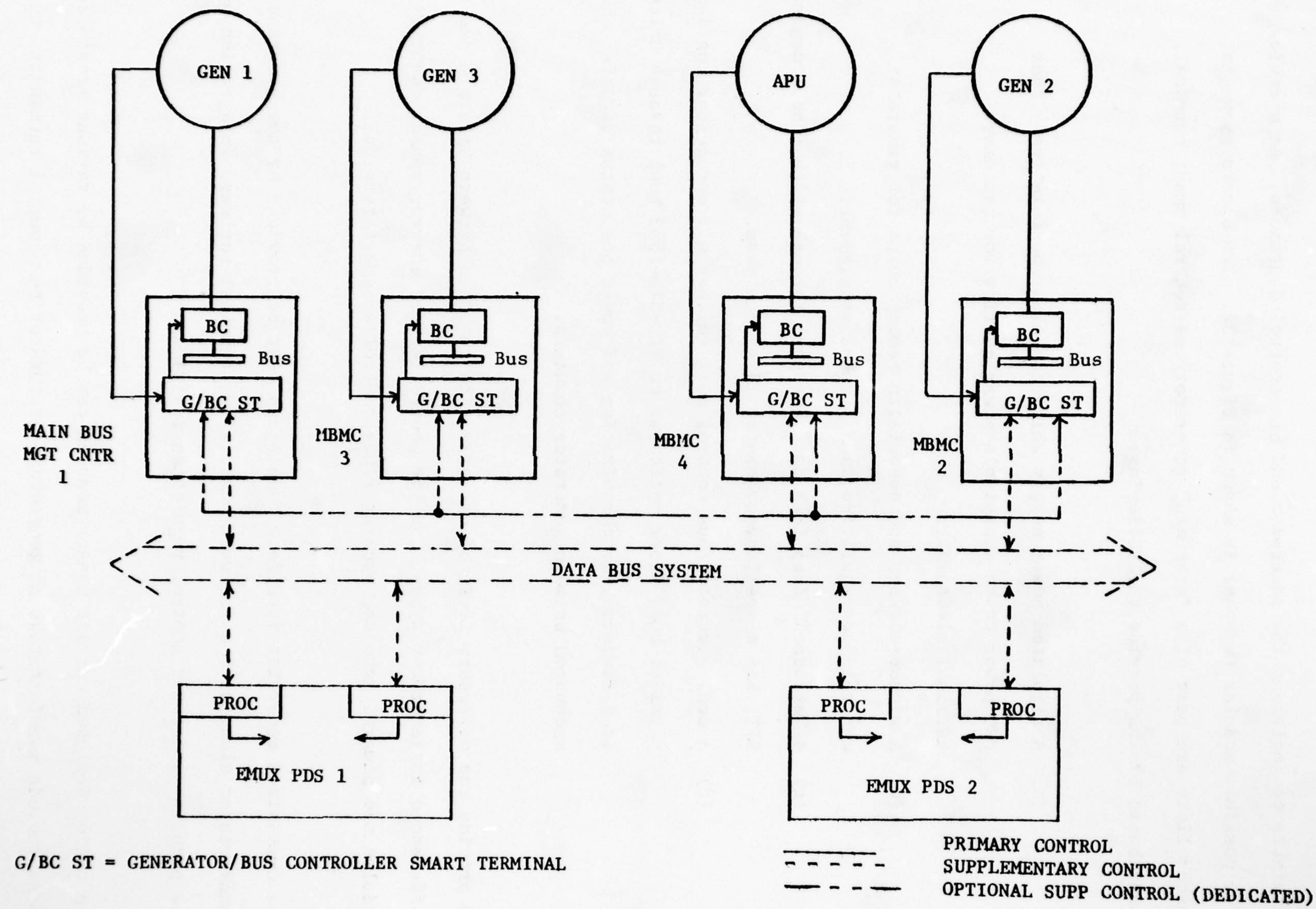

FIGURE 31 EMUX/GENERATOR CONTROL - CONCEPT 4

terminals to implement the generator and bus control functions. An overview of a possible modular terminal is shown in Figure 32. The blocks shown in dashed lines are part of a "standard" generator/bus control smart terminal. These basic blocks provide the following:

(a) A regulated power supply which converts the relatively poor PMG power to an acceptable power quality for the smart terminal electornics.

(b) A microprocessor and associated memory chips for generator control and regulation, and for bus management.

(c) A "standard" dual EMUX data bus port through which load management, BIT, and miscellaneous control data can pass.

(d) A dual generator/bus control port (either microprocessor configured parallel bus, fiber optic bus or MIL-STD-1553 bus) through which load division, synchronization and power bus status data is exchanged between generator channels.

To provide the necessary level of customization required between aircraft due to differences in generator types, and bus controller and current sensor quantities; modules are plugged into the generator/bus controller smart terminal.

The customized generator interface module can best be provided by the generator manufacturer along with a software package for terminal software integration of the peculiar generator control/regulation functions.

The custom designed current sensor modules can be provided by various manufacturers. These include manufacturers of generators generator/bus control terminals, current

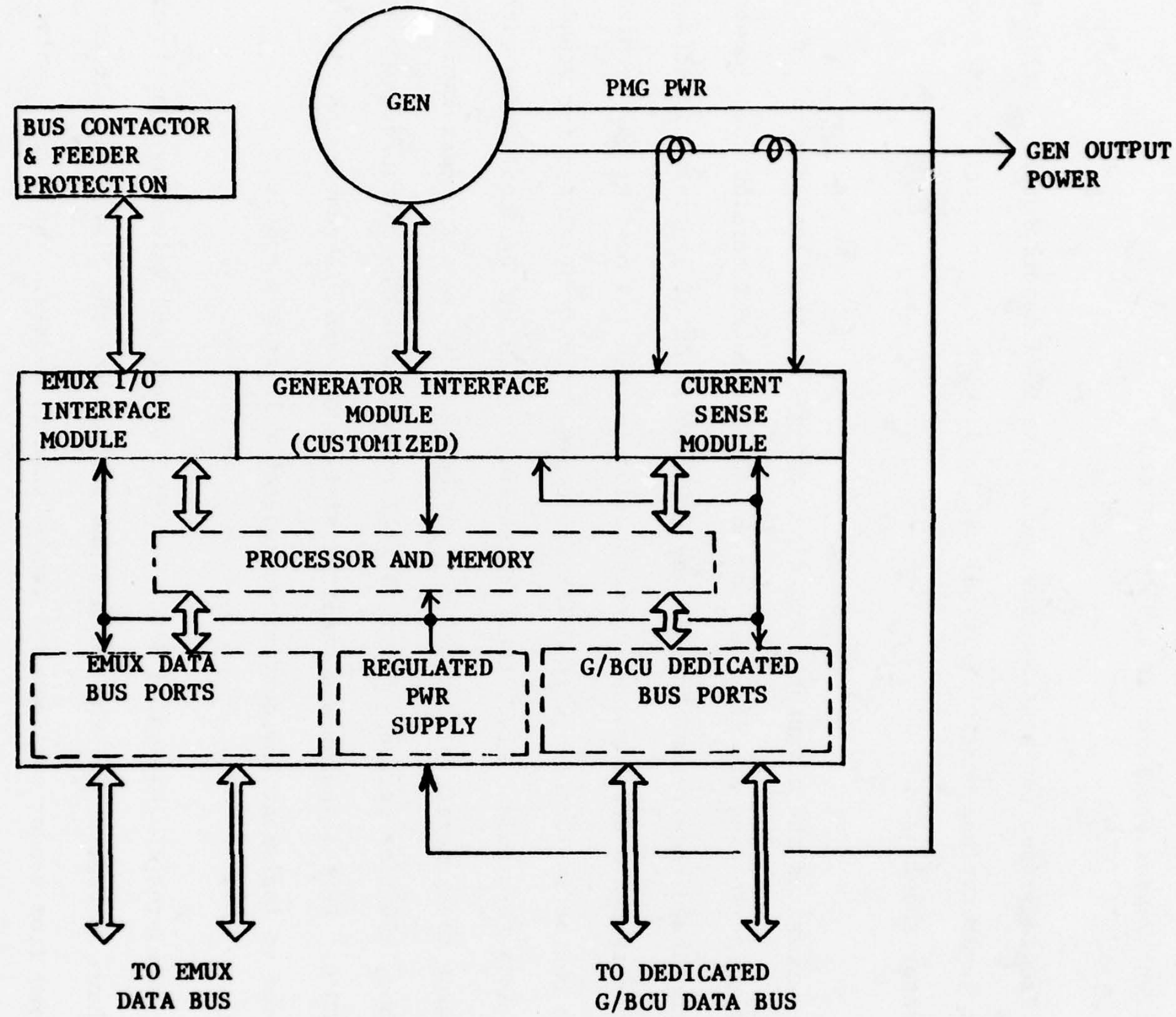

FIGURE 32 MODULAR GENERATOR/BUS CONTROL TERMINAL

sensors and aircraft. These modules could possibly be "standard" modules since current sensor operation is easily defined.

The EMUX I/O interface can best be provided by the EMUX manufacturer. Similarily, the basic generator/bus control terminal can be developed by either the EMUX or the generator supplier.

Table 21 summarizes the advantages and disadvantages of this last control concept. The major disadvantages are the need for an additional EMUX terminal development and the continued requirement for A/D and D/A conversion of generator interface signals. Although a new terminal configuration is required most of the terminal functions can be implemented with MSI/LSI circuits already developed for standard EMUX and AMUX terminals. Also, the requirement for A/D and D/A conversion circuitry to implement the generator interface may not be new. It is not unreasonable to expect that 1990 time period generators will be controlled by microprocessor implemented GCU's. If this becomes a reality, similar A/D and D/A conversion circuitry can be used to implement the customized generator interface module.

Table 22 tabulates the estimated G/BCU terminal weight and volume by functional blocks shown in Figure 32. The weights and volumes for the listed circuit cards are derived from vendor catalogs and vendor submitted data. The power supply parameters are derived from weight/volume rules-of-thumb for DC-DC converters. Finally, the weight and volume for housing, cooling, external connectors, etc., was derived by determining package size required to enclose the listed electronic components.

Table 21

EMUX/GENERATOR CONTROL - CONCEPT 4

ADVANTAGES	DISADVANTAGES
1. Dedicated processing required for generator regulation and control will minimize response slowdown due to EMUX related functions. 2. Standard G/BC smart terminal can be used with specialized plug-in modules provided by generator supplier for A/D conversion of generator functions. 3. Not limited to standard EMUX terminal I/O configuration i.e., analog or serial digital data can be accommodated. 4. Interface with EMUX bus provides access to BIT data and load management information. 5. Dedicated data bus between G/BC smart terminals yields fast channel-channel coordination while not requiring modification of EMUX architecture/operation. 6. PMG power required from each generator is limited to approximately 30 watts, which is within the present PMG capacity. 7. Critical loads can be powered (uncontrolled) even without EMUX operation.	1. New development required for generator/bus control smart terminal. 2. Smart terminal software coordination required between generator supplier and airframe (systems) contractor. 3. A/D conversion required for generator and current sensor data if not already digital. 4. Increase in total data bus quantities required to achieve desired response.

TABLE 22

WEIGHT/VOLUME ESTIMATE
FOR G/BCU TERMINAL

ITEM	VOLUME (IN^3)	WEIGHT (LBS)
BASIC TERMINAL		
o EMUX DATA BUS PORTS	20	0.5
o MICROPROCESSOR AND MEMORY	34	0.85
o G/BCU DATA BUS PORTS	16.5	0.30
o POWER SUPPLY (DC-DC) ~66 WATTS	33	1.0
EMUX I/O INTERFACE MODULE	20	0.5
CURRENT SENSOR MODULE	12	0.3
GENERATOR INTERFACE MODULE	108	2.7
SUB-TOTAL	243.5	6.15
MISCELLANEOUS HOUSING, COOLING, EXTERNAL CONNECTORS, ETC.	130	6.5
TOTAL (EST)	374	12.65

The estimated terminal weight of 12.6 pounds is reasonable when compared to dedicated conventional GCU typical weights of 3 to 12 pounds. (The higher GCU weights apply to units with BIT capability.) The G/BCU terminal not only provides GCU functions but also controls bus contactors and feeder protectors/controllers as well as providing a communication link to the EMUX system.

For generator designs which incorporate the GCU function within the main generator housing (e.g., the F-18 VSCF generator), the G/BCU concept can still be maintained. For this case, the generator enclosed GCU can include data bus ports and a small EMUX I/O (discrete) interface module. The remaining modules shown in Figure 32 can be contained in the generator GCU.

Due to the complex interface between generator (and converter) and the associated control/protection circuits, there are advantages to installing the GCU within the generator/converter package. The major advantage lies in reduction of external interface harnessing and associated EMI/RFI problems.

The four control concepts are qualitatively compared in Table 23 to identify those concepts for which additional study is warranted. As indicated in the table, only the first and fourth concepts are particularly feasible. The only significant differences between these two concepts are:

(a) Concept 1 may be implemented without a microprocessor while concept 4 will definitely need a microprocessor.

(b) Concept 4 includes provisions at the generator control unit for direct EMUX bus interface. Concept 1 requires data transfer through an EMUX remote terminal.

TABLE 23

COMPARISON OF GENERATOR CONTROL CONCEPTS

PARAMETER	CONCEPT			
	(1) CONVENTIONAL	(2) STANDARD EMUX	(3) EMUX SMART TERM	(4) GCU SMART TERMINAL
Feasibility	High - System Presently Exists	Low - Not likely to be Implemented	Low - Standard Hardware not Compatible	High - Within State-of-the-Art
System Response	Sufficient	Too slow	Likely too low	Marginal
Interface Compatibility	High (Exists)	-	Low - A/D Conversion Required	Medium - A/D Conversion Desirable
Flexibility to Performance Adaptation	Low Unless Microprocessor Implemented	-	High - Programmable Performance	High - Programmable Performance
Weight	Low	-	Moderate	Low to Moderate
Built-in-Test	Difficult	-	Moderate	Low to Moderate
Reliability	High	-	Moderate	Moderate
Vulnerability	High	-	Moderate	Moderate
Technical Risk	Low	-	High	Moderate

If the future equivalent to a conventional GCU is implemented with a microprocessor for flexibility, BIT processing or whatever reason, then the approach defined for concept 4 should be pursued to standardize the interface with other aircraft systems.

Pursuant with this rationale, concepts 1 and 4 are recommended for the preliminary electrical system design. In addition, it is assumed that the G/BCU smart terminal can be contained within the generator housing with bus/feeder protector and load shedding software provided by the aircraft systems integrator.

5.6 AC BUS CONTROLLER STUDY

The AC bus controller function can be implemented with an electro-mechanical device, a solid state device or a hybrid (electromechanical and solid state) device. The advantage and disadvantage of each implementation is given in Table 4.

The main advantages of an all solid state bus controller are the relatively fast response during turn-on and turn-off, and the ability of the device to be turned on and off at the zero axis of the sinewave. Both of these advantages contribute to improved power quality. The fast turn-on and turn-off response allows a power bus to be transferred from one power source to another in typically 5 milliseconds (one cycle for turn-on and one cycle for turn-off). Electromechanical bus controllers require typically 30 milliseconds for bus transfer (10 milliseconds for turn-on and 20 milliseconds for turn-off). The hybrid device requires typically 22 milliseconds (one cycle turn-on and 20 milliseconds turn-off).

TABLE 24

AC BUS CONTROLLER CHARACTERISTICS

CONCEPT	ADVANTAGES	DISADVANTAGES
Electromechanical	1. Low voltage drop (low dissipation) 2. Small size and low weight 3. Low cost 4. Off-the-shelf availability 5. Excellent electric isolation	1. Moderate operating life 2. High EMI, contact bounce and chatter 3. High control power (15 watts typical) 4. Slow response - $\left.\begin{array}{l}\text{Turn-on = 20 ms}\\ \text{Turn-off = 10 ms}\end{array}\right\}$ typical
Solid State	1. Fast response - Turn-on = 1.0 ms Turn-off = 2.5 ms 2. Low EMI - Turn-on and turn-off at zero crossover of sinewave 3. Long operating life over 1,000,000 cycles 4. High reliability - no moving parts 5. Low control power - typically 1.0 W 6. Excellent shock and vibration characteristics (no contact chatter)	1. High voltage drop - typically 1.5 volts (high pwr. dissipation) 2. Cooling required 3. A failure mode can result in half wave operation, i.e., DC power to load bus 4. Costly due to large SCR's required to accommodate fault current 5. Poor electric isolation typically 5.0 MA leakage 6. Large and heavy (including sink provisions)

TABLE 24 (Continued)

AC BUS CONTROLLER CHARACTERISITCS

CONCEPT	ADVANTAGES	DISADVANTAGES
Hybrid	1. Low voltage drop (low power dissipation) 2. Fast turn-on response, typically 1.0 ms 3. Low EMI - Turn-on and turn-off at zero crossover of sinewave 4. Long operating life typically 1,000,000 cycles 5. Small and low weight 6. No contact bounce or chatter	1. High cost due to redundant switches 2. A failure mode can result in half wave operation (DC power to bus) 3. Poor electric isolation (typical 5.0 MA leakage) 4. Reduced reliability due to added parts 5. High control power - typically 15 watts 6. Slow turn-off response - typically 20 milliseconds

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Both the solid state and hybrid concepts have the ability of switching at the zero axis of the sinewave. This capability reduces the switching EMI to a minimum. Furthermore, no contact bounce is experienced which further reduces EMI over the electromechanical device.

Voltage surges resulting from normal load switching are caused by the inherent regulation of the power source. Bus controller switching characteristics has little, if any, impact on the voltage surge characteristics, consequently, all three bus controller implementation concepts are acceptable.

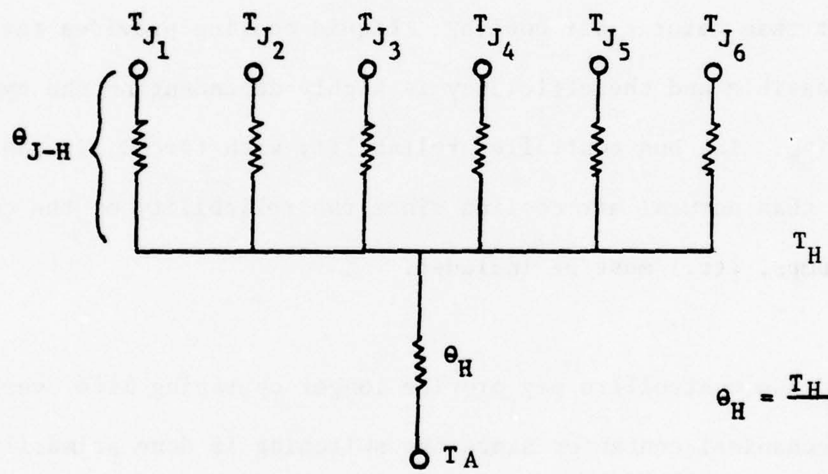
Fault clearing is accomplished best by an all solid state device because the turn-off response is fast. Fault currents below a specified value can be interrupted within one cycle (2.5 ms). This is not possible with either the electromechanical or hybrid concepts.

The over-riding disadvantages of solid state bus controllers is the high power dissipation resulting from the semiconductor voltage drop. This dissipation coupled with the relatively low temperature rating of semiconductor devices (125°C for SCR's) makes cooling an important factor since the inherent reliability of semiconductor devices depends a great deal on using the device within its stated thermal limits. The most reliable cooling means is simply a heat sink or heat dissipator with natural air cooling. However, this mode of cooling is feasible, from a size and weight standpoint, only if the dissipation is low. Table 25 gives the current values per phase of various generating system ratings. Also given are power dissipations resulting from a typical semiconductor voltage drop of 1.5 volts. The thermal resistance values shown in the table are the heat

TABLE 25

AC BUS CONTROLLER COOLING CONSIDERATIONS

GENERATOR SYSTEM RATING KVA	CONTROLLER CURRENT PER PHASE AMPS	CONTROLLER POWER DISSIPATION (WATTS)			HEAT SINK THERMAL RESISTANCE °C/W (ΘH)
		PER SCR	PER PHASE	TOTAL	
40	116	87	174	522	0.055
60	174	131	261	783	0.037
90	261	196	391	1173	0.025
150	435	326	652	1956	0.015



$$\Theta_H = \frac{T_H - T_A}{W}$$

THERMAL RESISTANCE CIRCUIT

Θ_H = Thermal resistance of heat sink

T_H = Temperature of heat sink (100°C)

T_A = Temperature of ambient air (71°C)

W = Controller dissipation (total)

T_J = Temperature of semiconductor (125°C)

Θ_{J-H} = Termal resistance from junction to heat sink

sink maximum values required to maintain the semiconductor temperature below its rating when operating in an ambient air of 71^{o}C with natural air cooling. Figure 33 is a guide to the volume (surface area) of heat sink that is required to meet a given thermal resistance value and is based on an efficient heat sink design. An efficient design for natural air cooling or forced air cooling consists of a sink with extruded fins. Figure 33 clearly illustrates that below 0.3^{o}C/W, very large increases of volume are necessary for small increments in thermal performance. The implication is that some means other than natural air cooling is required to reduce the volume. Other means include forced air cooling and liquid cooling. Forced air cooling is typically three times more efficient than natural air cooling. Liquid cooling provides the most compact design possible and the efficiency is highly dependent on the type of liquid used for cooling. The bus controller reliability with forced air and liquid cooling is lower than natural air cooling since the reliability of the cooling hardware (fans, pumps, etc.) must be included.

Hybrid AC bus controllers may provide longer operating life over that of an electromechanical contactor since the switching is done primarily by the solid state device. Also, fast turn-on time is provided but turn-off time is longer since the time required to open the contactor is added to the time required to open the solid state switch. The primary disadvantages of the hybrid device is the higher cost and the reduced reliability due to the added components. The longer operating life is not a significant advantage in the bus controller application since 100,000 cycles (typical rating of an electromechanical contactor) represent approximately 60 years of aircraft life.

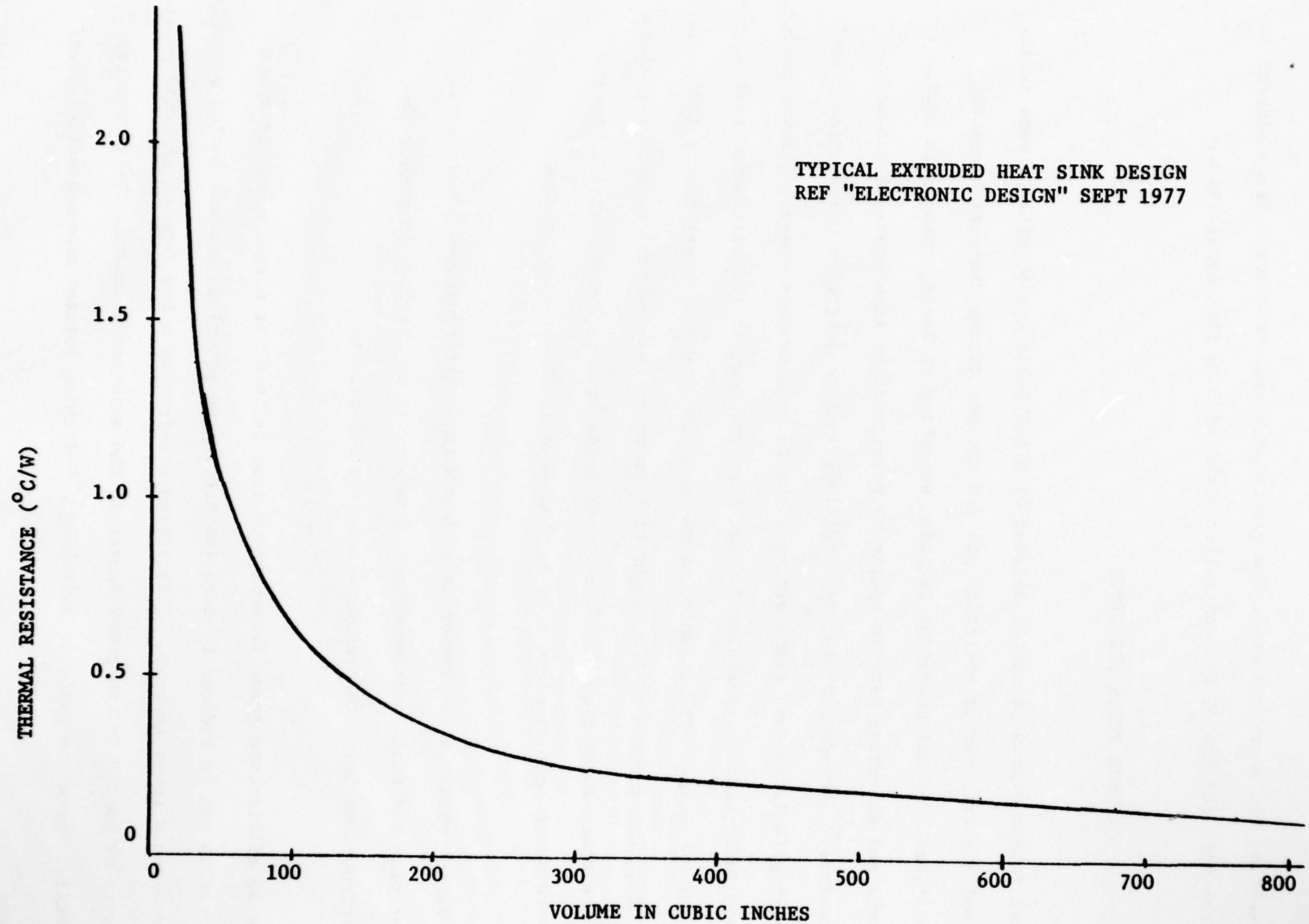


FIGURE 33 THERMAL RESISTANCE VS HEAT SINK VOLUME FOR NATURAL AIR COOLING

Because of the above concerns, the electromechanical contactor is considered to be the best for the AC bus controller (LINE AND BUS TIE) application.

5.7 GAPLESS POWER BUS STUDY

Power interruptions presently defined by MIL-STD-704 are 50 milliseconds during normal electric system operation and 7.0 seconds during abnormal operating conditions. Normal operation includes switching of loads, changes in engine speeds and switching between operating power sources (synchronized and unsynchronized). Abnormal operating conditions include generator malfunctions, fault clearing/isolation and transferring a bus to an alternate operating power source. Power interruptions exceeding 7.0 seconds can occur if the alternate power source is not operating. For example, start-up of an alternate power source (APU) can be under the control of the flight crew in which case several seconds may elapse before power is available. These power interruptions become critical to the operation of some equpment such as that used in fly-by-wire systems.

It is not possible to provide an AC bus with no interruptions covering all operating and fault contingencies. However, it is possible to reduce the interruptions below the values allowed by MIL-STD-704.

The 50 millisecond power interruption time defined for normal electric system operation can be reduced to zero (uninterrupted power) in systems having operating synchronous power sources. This includes switching a bus from an external power source or an APU driven power source to the main power source. This requires parallel operation prior to switching. Switching between unsynchronized power

sources can be accomplished within the 20 millisecond goal with the use of solid state or hybrid bus controllers. These components, however, are not practical at high current levels as was noted in paragraph 4.1.6.

The real problem of providing gapless power occurs during fault conditions. Existing components used to provide fault clearning and fault isolation include thermally actuated devices such as time delay fuzes, circuit breakers and electro-mechanical contactors. It is not possible to meet the 20 millisecond goal with these components although the time can be reduced substantially from the MIL-STD-704 limits with the use of instantaneous trip magnetic breakers and/or rapid fault clearning fuzes operating in conjunction with special electromechanical contactors. Conventional contactors have typical dropout times of 10 milliseconds with DC coils and 40 milliseconds with AC coils. The relatively long drop-out time with AC coils is due to the "slugging effect" of rectifiers used for converting the AC voltage to DC voltage. A design technique can be employed which will eliminate the "slugging effect" and thereby decrease the drop-out time equivalent to that of a DC coil. This design technique typically consists of a semiconductor switch located between the rectifiers and the coil. The possibility of meeting the 20 millisecond time duration becomes more feasible if a solid state bus controller is used in lieu of the electromechanical contactor. Closing and opening time for these devices is typically 2.5 milliseconds respectively. Solid state and hybrid bus controllers are practical in applications requiring relatively low current switching, i.e., a bus of limited capacity. Typically, the limited capacity bus serves power only to those loads which are sensititive to power interruptions. Minimum power interruption is accomplished by transferring the bus to a battery powered inverter. The transfer device is either a solid state or a hybrid bus controller.

A technique for minimizing the possibility of a power loss is to have utilization equipment supplied power from a multiple of sources. The equipment contains redundant power supplies with their outputs diode isolated. This technique does complicate the design of the utilization equipment and requires additional power distribution circuits.

5.8 POWER DISTRIBUTION AND CONTROL STUDY

The Power Distribution and Control Subsystem (PDS) for an advanced aircraft is projected as being centered around the EMUX power control concept. The EMUX implementation is envisioned to exist for all but the very simplest of electrical systems. The "breakeven" point for selecting EMUX over the conventional electrical system architectures varies depending upon life cycle cost contributions from maintainability, reliability, weight and production costs. Figures 34, 35, 36 and 37 illustrate gross trade-offs for mission failure rate, maintenance rate, production costs and weight respectively.

Figure 34 indicates a mission failure rate breakeven point of less than one circuit for preference of EMUX over the conventional PDS. The very low EMUX failure rate is due predominately to the EMUX redundancy and to the low solid-state switchgear failure rates. The failure rates of 0.024 shown in the associated breakeven equation accounts for processor and data bus coupler mission reliabilities.

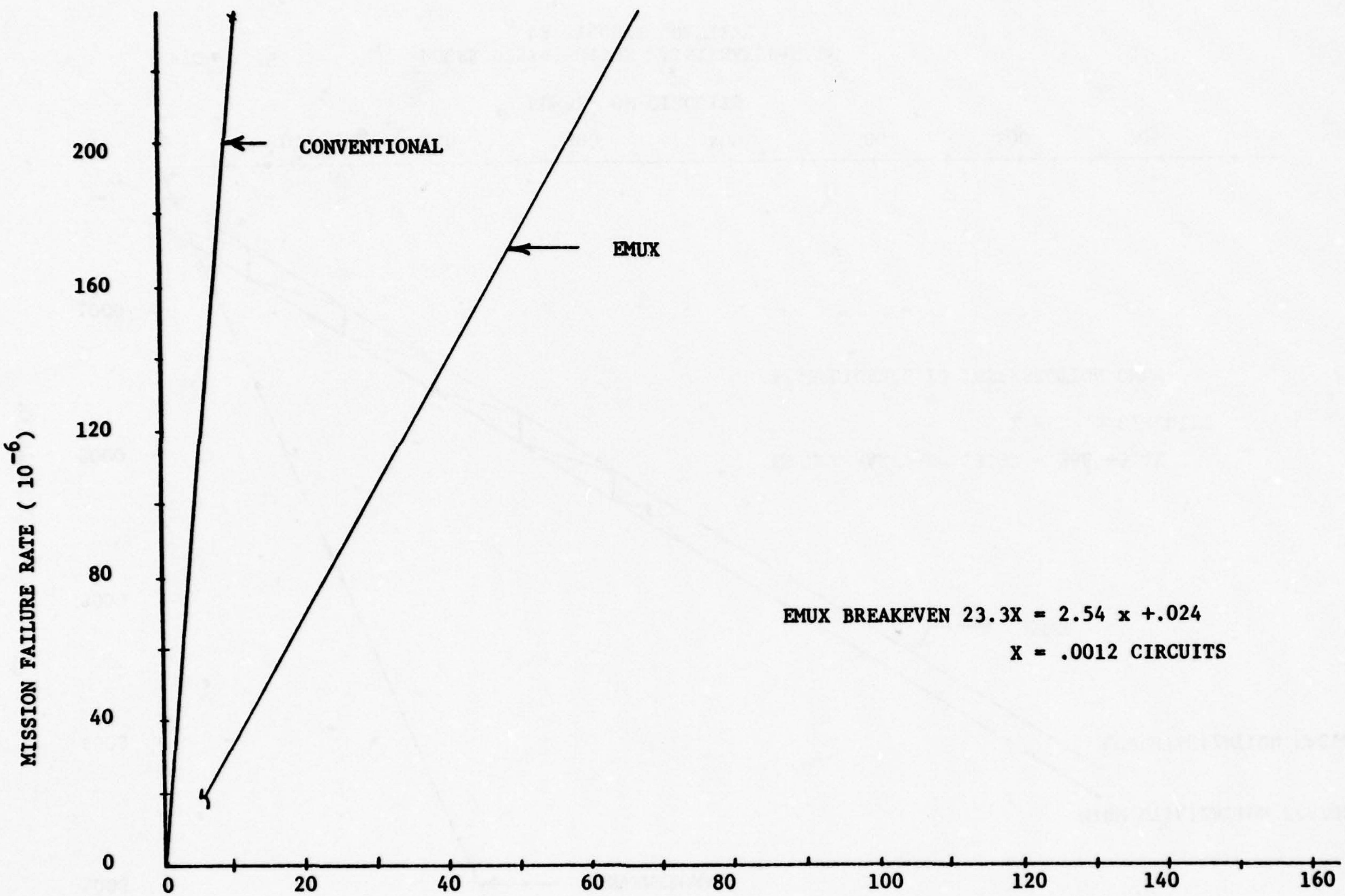


FIGURE 34 POWER DISTRIBUTION MISSION FAILURE RATE VS CIRCUIT QUANTITY

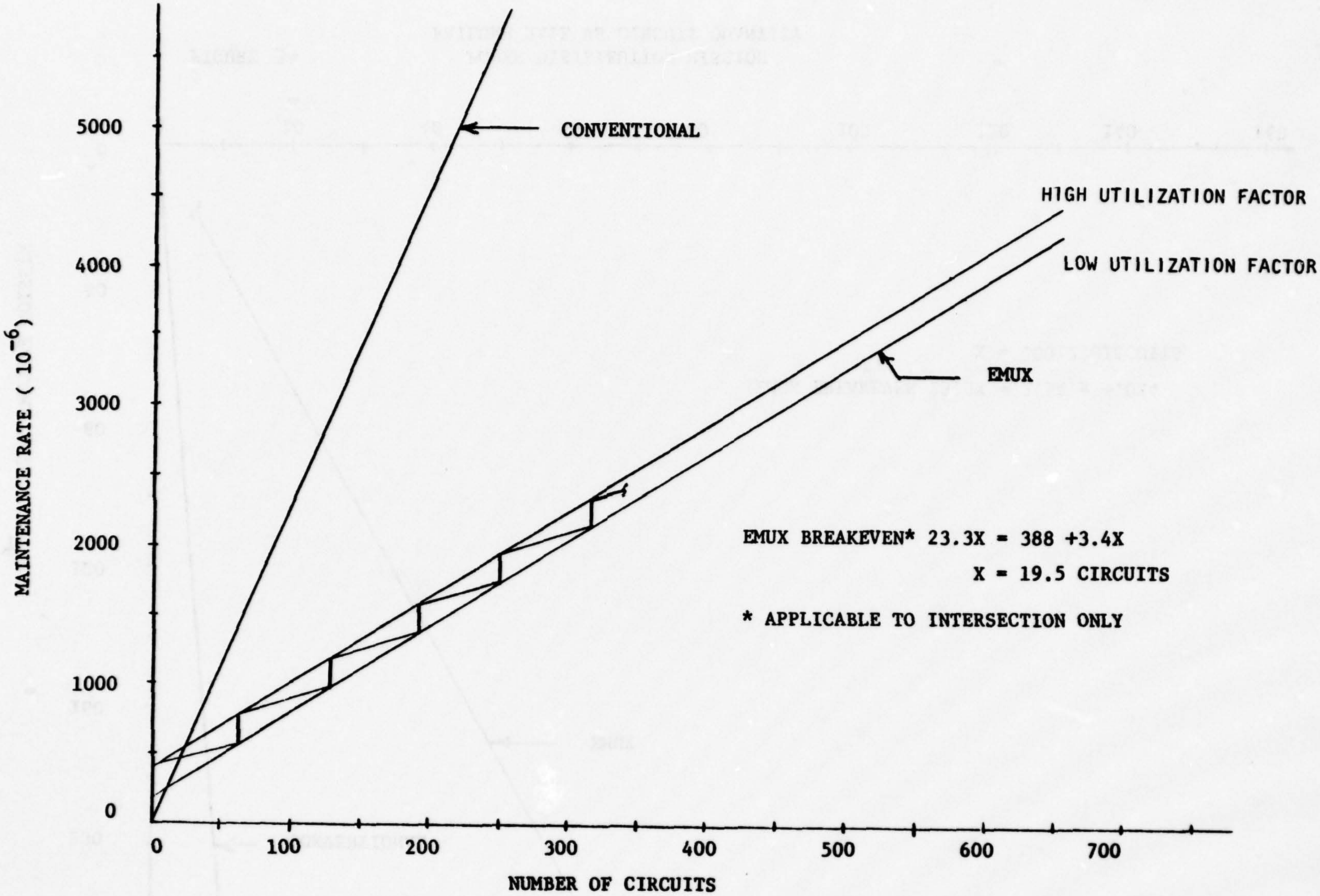


FIGURE 35 POWER DISTRIBUTION MAINTENANCE RATE VS CIRCUIT QUANTITY

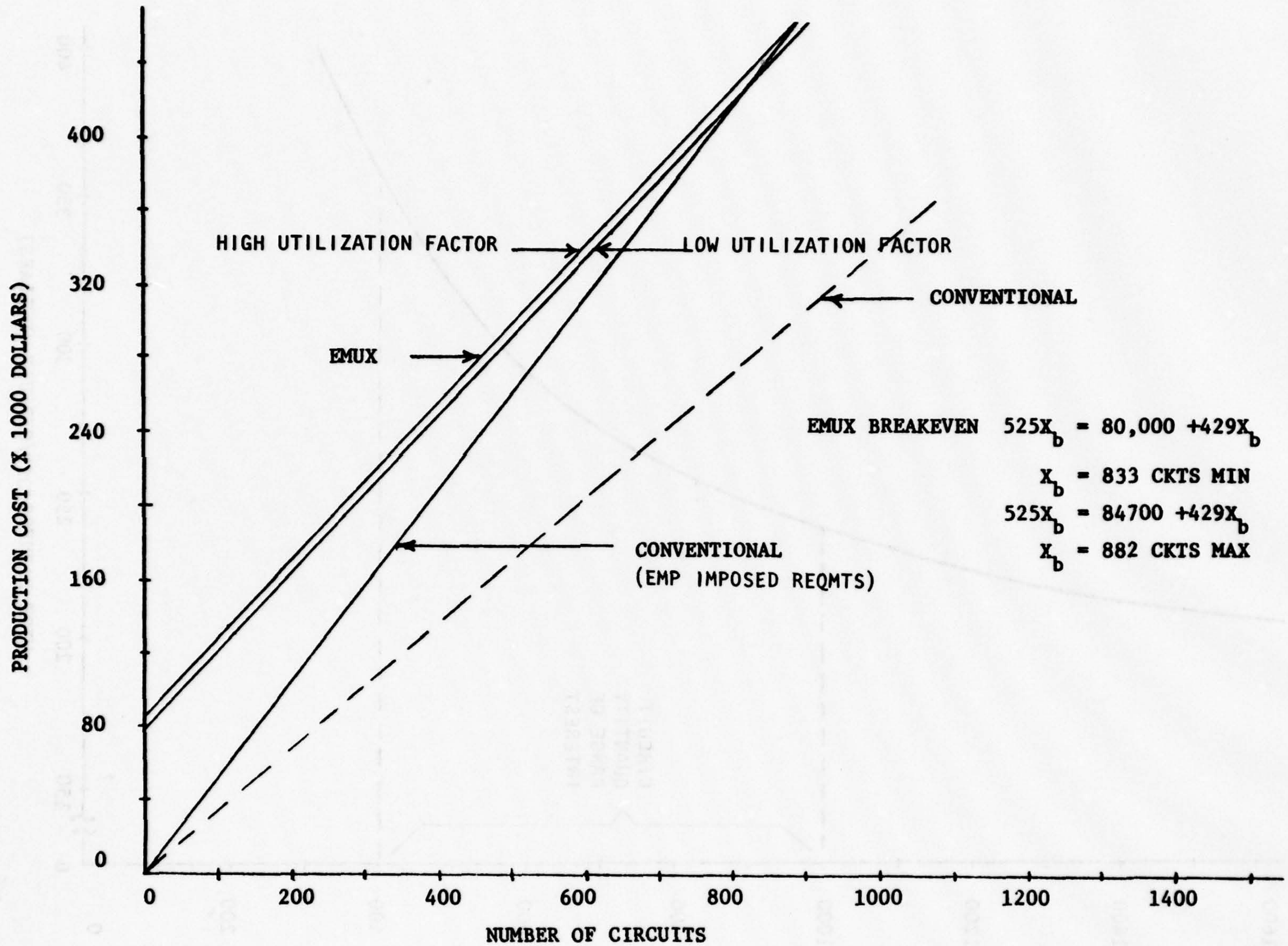


FIGURE 36 PRODUCTION COST VS CIRCUIT QUANTITY

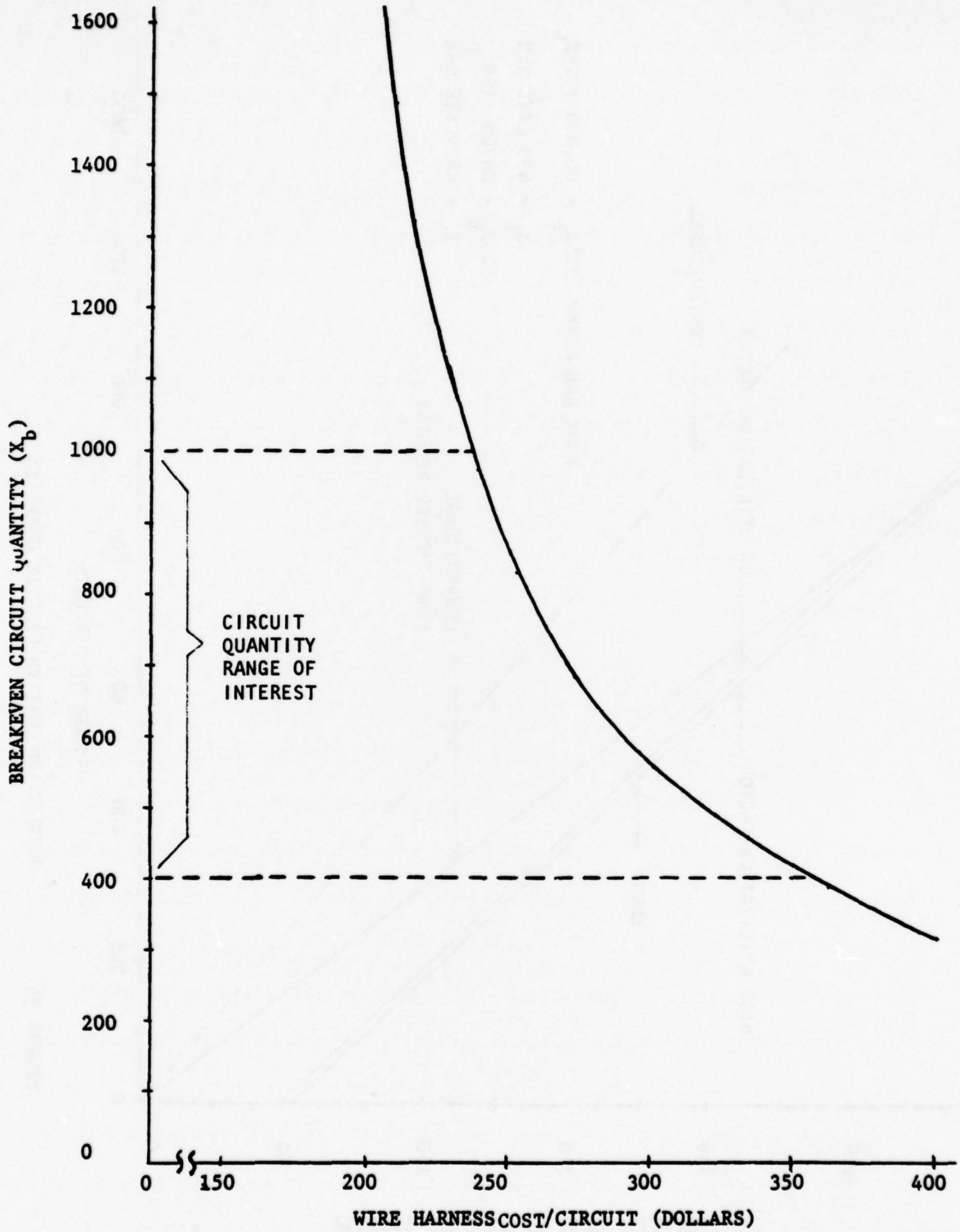


FIGURE 37 PRODUCTION COST BREAKEVEN SENSITIVITY OF EMUX VS CONVENTIONAL SYSTEMS

The maintenance rate for the two PDS concepts is shown in Figure 35 as a function of the number of aircraft circuits. The "staircase" function depicted for the EMUX option results from installation of integral quantities of EMUX remote terminals. These maintenance curves indicate a breakeven point of 20 circuits for preference of EMUX over conventional PDS implementation based solely on maintenance rate. A secondary benefit of EMUX with respect to maintenance activities results from the extensive BIT available for the PDS. The BIT provided by EMUX will significantly improve the manhours/maintenance action for an EMUX based PDS as compared to a conventional PDS. Reference 2 indicated a 27 to 31 percent reduction in average manhours per maintenance action for an EMUX based PDS.

The next life cycle cost factor comparison plot is shown in Figure 36. This figure illustrates projected production costs for the two PDS options. As shown, the EMUX option breakeven occurs between 833 to 882 circuits. The confidence factor for this number is very low, however. The low confidence is due to the high breakeven sensitivity to the difference wire harness costs between conventional and EMUX systems. Figure 37 shows the variation in the breakeven circuit quantity as a function of the harness cost delta between the two PDS concepts. As indicated, the breakeven point is very sensitive to the per circuit harness cost delta in the circuit quantity range of interest. The most important comment that can be made on the plotted data is that unless the conventional system harness cost (per circuit) is at least 235 dollars more expensive than the EMUX system harness cost; the EMUX production cost will be higher than the conventional system. It is not expected that this cost delta would occur on a typical fighter/attack aircraft unless extremely stringent EMP requirements were imposed on the electrical harness.

The dashed line in Figure 36 represents the expected production cost relationship for conventional aircraft with present day design requirements while the solid line represents conventional aircraft with stringent EMP requirements.

The final life cycle cost contributor relationship is illustrated in Figure 38. This figure plots estimated power distribution weight as a function of the total electrical system input and output quantities and were derived from reference 1. The diagonal lines in the figure represent linearized weight growth for EMUX and conventional PDS concepts in aircraft which make extensive use of Avionic Multiplexing (AMUX).

The relationship between PDS complexity and weight is more clearly visualized when comparing weight savings . The weight savings are shown in Figures 38, 39 and 40 (percentage) and indicate, for example, that an EMUX PDS for a four engine aircraft would yield an approximate 50 percent weight savings over a conventional PDS.

It is possible to assign a dollar value to weight savings for a specific aircraft application. However, since this value is very sensitive to such factors as total weapon system weight, specific fuel consumptions, mission profile (e.g., loiter time, distance to target, climb/descent rates), etc., discussion of specific weight costs for a "typical" single engine and four engine aircraft would have limited value. Recent weight cost values which have appeared in various electrical/ electronic system study reports for single and twin engine aircraft range from 300 to 1,200 dollars/pound over the life of one air vehicle. However, a likely more significant use of the saved PDS weight is to permit an increase in the amount of

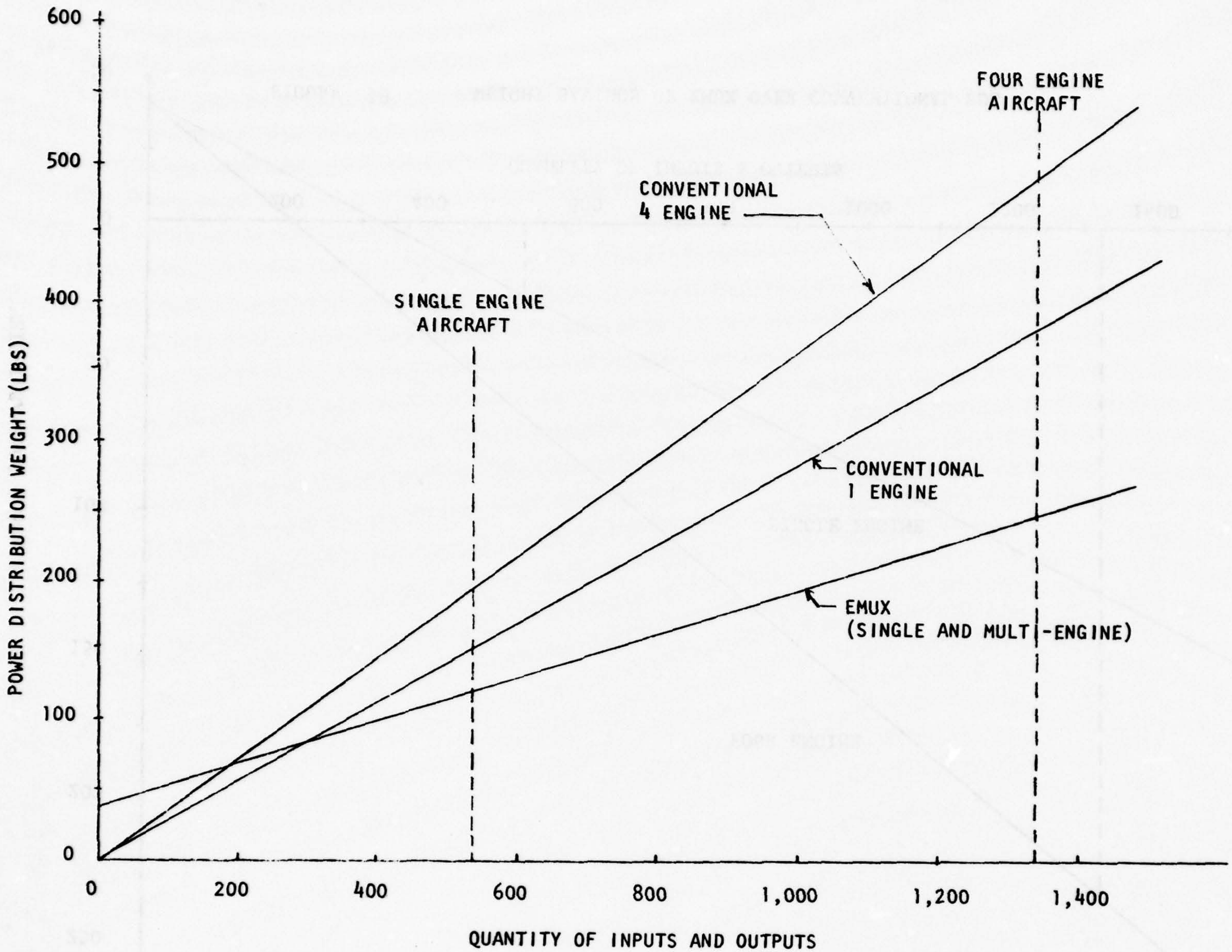


FIGURE 38 POWER DISTRIBUTION WEIGHT VS SYSTEM COMPLEXITY

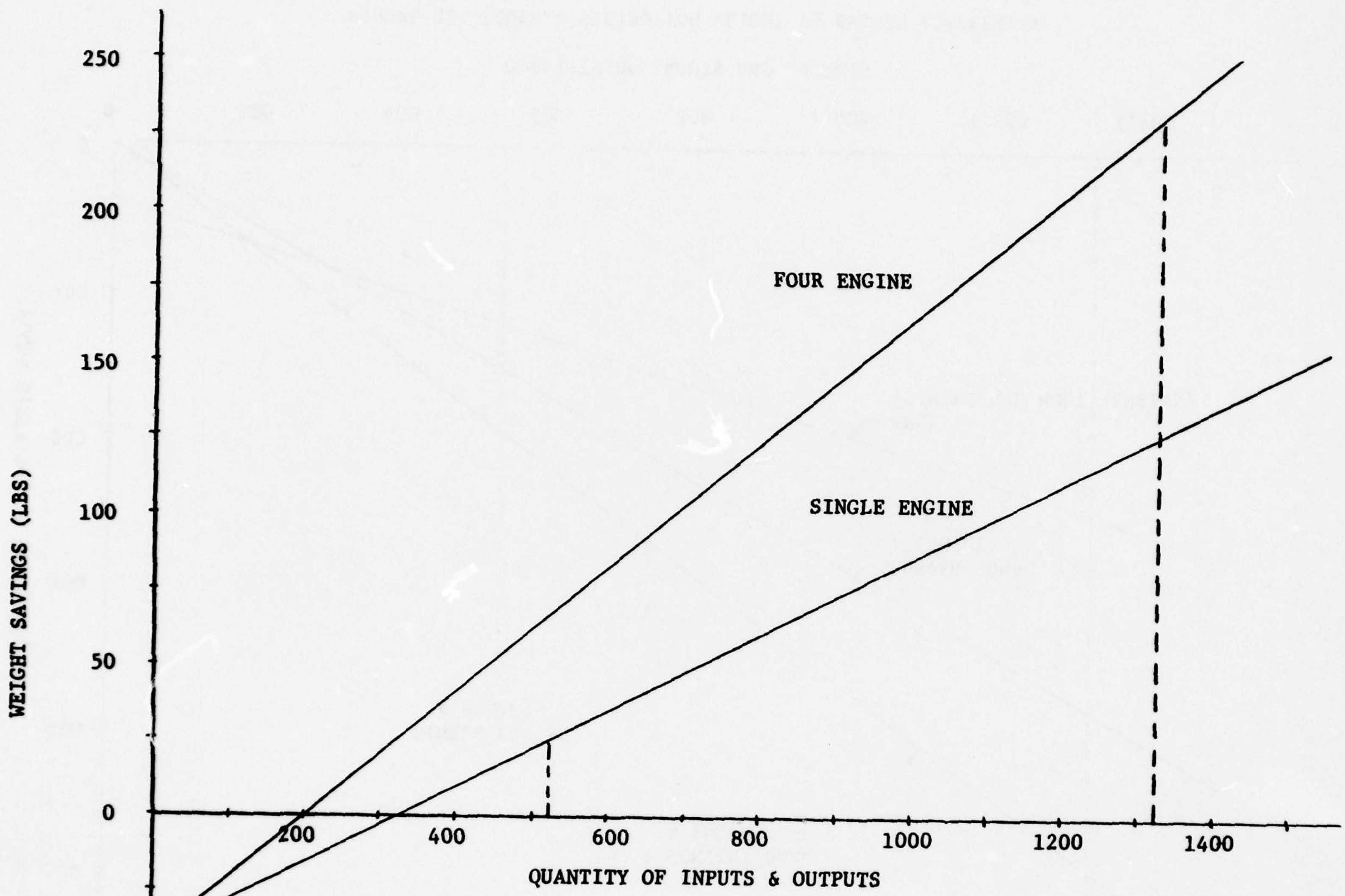


FIGURE 39 WEIGHT SAVINGS OF EMUX OVER CONVENTIONAL PDS

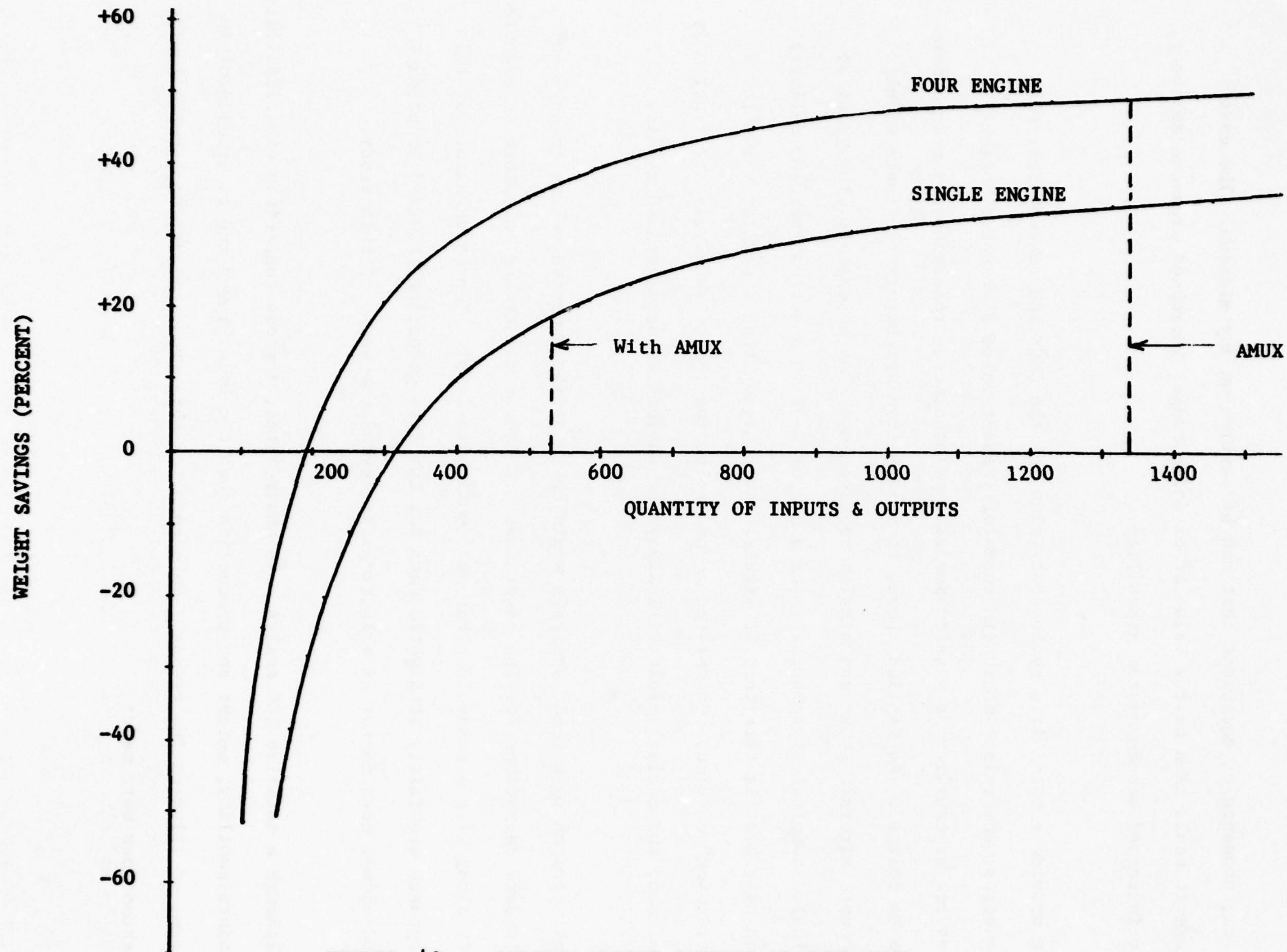


FIGURE 40 PERCENT WEIGHT SAVINGS OF EMUX OVER CONVENTIONAL PDS

fuel, ordnance or equipment that can be carried on any mission. The cost benefit would then be the value of extended range, increased ordnance delivery, or increased weapon system capability.

To provide a total life cycle cost value for the EMUX and conventional PDS concepts, the relationship (in terms of present value discounted dollars) between probability of mission success, maintenance, production and weight need to be defined. As implied above, these relative weightings fluctuate widely between aircraft types and mission requirements. As an example, the value of mission completion probability for a B-1 or B-52 on a nuclear weapons delivery run over hostile territory is significantly higher than a similar value for a "training" mission. Unless the ground rules for the LCC analysis are rigorously defined, the dollar result will always be subject to accuracy and debate.

One approach to the LCC analysis would be to define expected values (means) and standard deviations for the respective LCC factor values and calculate an expected LCC along with a standard error and confidence level. The calculations of this approach are fairly straightforward but the data gathering required to produce individual cost factor distributions is beyond the scope of this study.

Although a detailed LCC analysis is unwarranted, a gross comparison of reliability, maintainability, weight and production cost can be achieved and is meaningful in terms of illustrating trends. To accomplish this gross comparison, the following assumptions were made:

- o Cost of maintenance manhours = \$16/hour
- o Average manhours/maintenance action = 2.5 for conventional

  = 1.825 for EMUX
- o Flight hours/month/aircraft = 35
- o Aircraft life = 10 years
- o LCC cost sensitivity to weight = \$750/pound
- o Cost of mission abort = \$7500

A composite cost (C_T) can then be derived by weighting each of the various cost contributors considered. The composite cost equation is of the form:

$$C_T = K_R \lambda + K_w W + K_M MR + K_p PC$$

where: λ = failures (mission)/10^6 hours

W = weight in pounds

MR = maintenance actions (MA)/10^6 hours

PC = production cost

and weighting factors:

.K_R = 3.024×10^7 \$/(failure/10^6 hours)

.K_W = 750 \$/pound

.K_N = 1.23×10^5 \$/(MA/10^6 hours) - EMUX PDS

= 1.68×10^5 \$/(MA/10^6 hours) - Conventional PDS

.K_p = 1

From Figures 34, 35, 36 and 38, a compoiste cost equation can be derived for an EMUX and conventional system. These equations (as a function of the number of circuits) are:

$$C_{T(EMUX)} = 3.024 \times 10^7 (2.54 x + .024) + 750 (.308 \times 39.5)$$
$$+ 1.23 \times 10^5 (3.4 \times 388) + 1 (429 x + 8 \times 10^4)$$
$$= 7.722 \times 10^7 x + 4.856 \times 10^7$$
$$C_{T(CONV)} = 3.024 \times 10^7 (23.3X) + 750 (.647X) + 1.68 \times 10^5 (23.3X)$$
$$+ 1 (525X)$$
$$= 70.851 \times 10^7 X$$

Equating the two composite costs indicate that an EMUX based system would be "cheaper" in all instances.

Closer examination of the two composite cost equations result in the following observations.

(1) For ultra-simple systems (less than 5 circuits) the "fixed" maintenance and reliability costs dominate the composite cost of the EMUX based system.

(2) Weight and production cost factors are totally dominated by the R&M cost factors over the entire range of system complexity

In a general, generic sense, EMUX encompasses computerized data processing and multiplexed data transfer. When implemented with remotely located multiplex/demultiplex terminals which communicate with central data processors, virtually all electrical system information is transferred as low energy digital data over common data bus(s) rather than as "high energy" electrical flows over dedicated wires. The high energy transfer requirement is therefore simplified. With EMUX delivery of useful power from the power bus to the load is a direct

and very short path routed controller for circuit protection and switching logic control. No intervening logic is required between the load controller and the load. This minimizes wire length, harness complexity and hardware weight while improving system flexibility. In addition, by eliminating high energy logic processing with relays, switches, etc., inherent reliability improvements are possible due to the reduction in the quantity of voltage stressed and thermal stressed logic switching devices (i.e., relays and switches).

The EMUX concept provides flexibility in implementing the power distribution switching/protection functions with either electromechanical or solid-state hardware. Table 26 compares the various characteristics of electromechanical and solid-state load controllers. The most important parameters are power dissipation since it affects system efficiency, failure rate, weight, volume and cost.

The solid state power controller reliability is very sensitive to the average exposed operating temperature. This relationship is shown in Figure 41. The diagonal lines in the figure define the relationship between the thermal resistance from SSPC case to ambient for various ambient temperatures and the impact on SSPC failure rate. For example, if the SSPC installation design is such that a 1°C/watt thermal resistance from SSPC to ambient air is provided, the SSPC MTBF (at an average ambient temperature of 71°C) will be approximately $.11 \times 10^{6}$ hours. This yields a failure rate of 9 failures/10^{6} hours.

Most present generation avionic systems are designed for a maximum ambient of 71°C (MIL-E-5400 Class 2). For this reason it is not reasonable to use 71°C

TABLE 26

COMPARISON* OF POWER SWITCHING/PROTECTION HARDWARE

	ELECTROMECHANICAL	SOLID STATE
Power Dissipation (Efficiency)	2.75 watts (0.995)	16 watts (0.975)
Failure Rate	20 failures/10^6 hrs	2.4 to 10 failures/ 10^6 hrs
Weight (Device only)	0.735 lbs.	0.188 lbs.
Volume (Device only)	15 in^3	4.1 in^3
Cost	#250	$50 - 250
Voltage Drop	0.50 volts	3.2 volts max @ -54°C 2.8 volts max @ 100°C
Turn-on Time	30 mx max	1.5 ms max
Turn-off Time	12 ms max	2.0 ms max
Rupture Current**	3600 amperes	>>3600 amperes
Current Limiting	Not practical	270 - 330% (optional)
Turn-on Energy	38 joules max	<< 1 joule
Overload	200%	>> 200%
Leakage Current	∿2μ amperes	>500μ amperes

*Comparison based on 5 ampere rated device operating at 80 percent rating.

Electromechanical data based on MIL-C-83383/20, Solid State data based on MIL-P-81653/4 as modified by NADC-30-TS-7602/3.

**Rupture current is that value of current in a circuit which reflects the capabilities of the power source without the effects of the power switch.

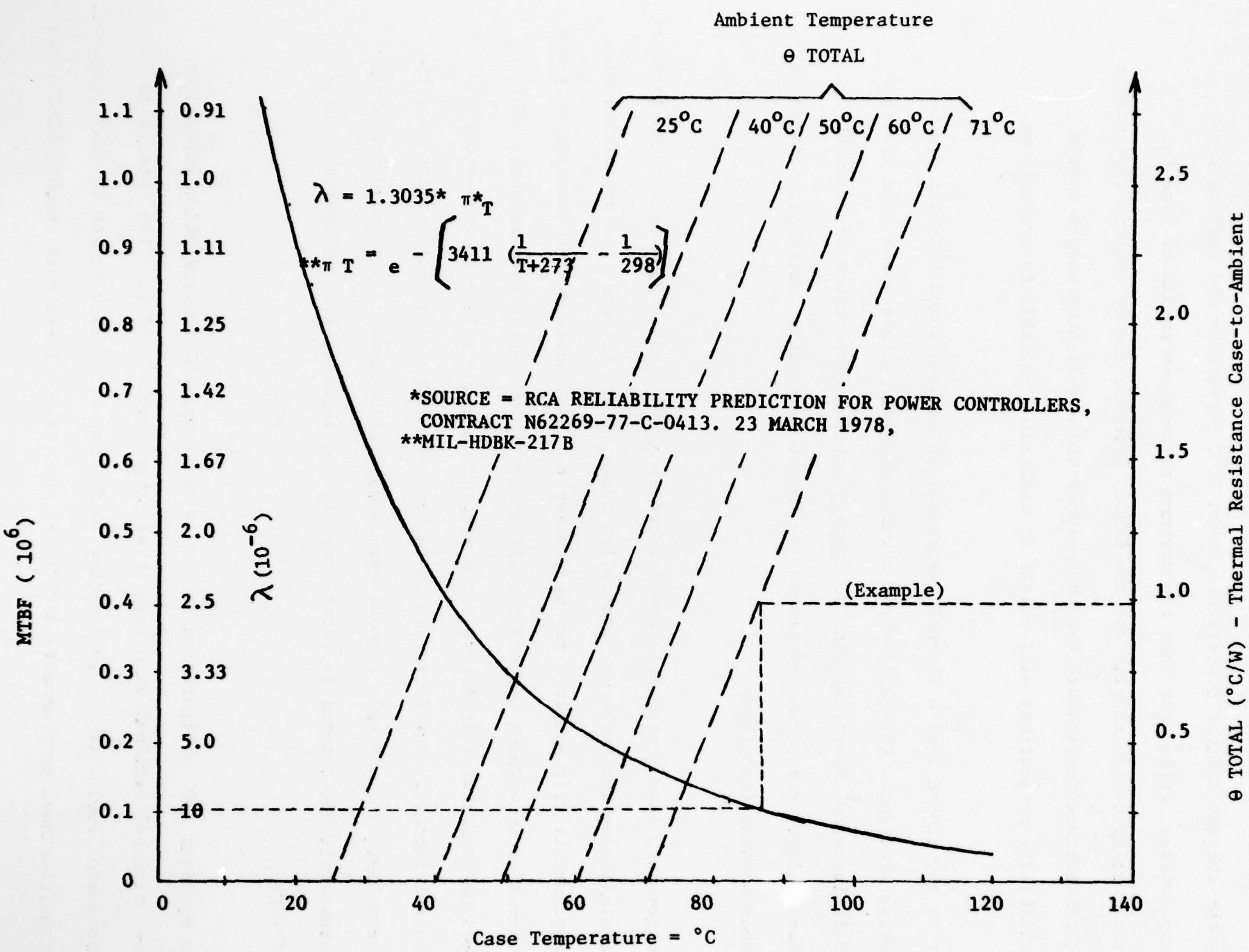


FIGURE 41 RELIABILITY PREDICTION FOR POWER CONTROLLERS

as the average ambient temperature, when $71^{\circ}C$ represents the maximum rated temperature. It is true that the maximum temperature should be used for establishing design margins for the SSPC. However, use of maximum temperature as the average temperature for the purpose of establishing weapon system reliability projections will result in misleading reliability projections.

It is envisioned that a thermal resistance of between 1 and $2^{\circ}C$/watt can be easily provided. This will yield a range of SSPC failure rates from 2.4 to 10 failures/10^6 hours assuming an average ambient temperature of 25 to $60^{\circ}C$. These failure rates can, of course, be improved by using forced air, liquid or vapor cooling techniques.

In general, load controllers are defined by MIL-P-81653. However, in the interest of minimizing the number of controller types, i.e., ratings, the use of hardwire programmable trip settings may be desirable. This would permit stocking only one controller type (part number) for all aircraft controller requirements. The optimum number of trip levels for a programmable controller is influenced by several factors. One factor is the number of channels (poles) selected for the controller, i.e., single channel versus multiple channel. The choice of single versus multiple channel is established by the selected load management center concept.

An optional LMC configuration to the conventional LMC configuration integrates the universal terminal with trip level programmable SSPC's. This integrated load management center (ILMC) is depicted in Figure 42. This approach is only practical if multichannel programmable SSPC's are available. These multichannel SSPC's can

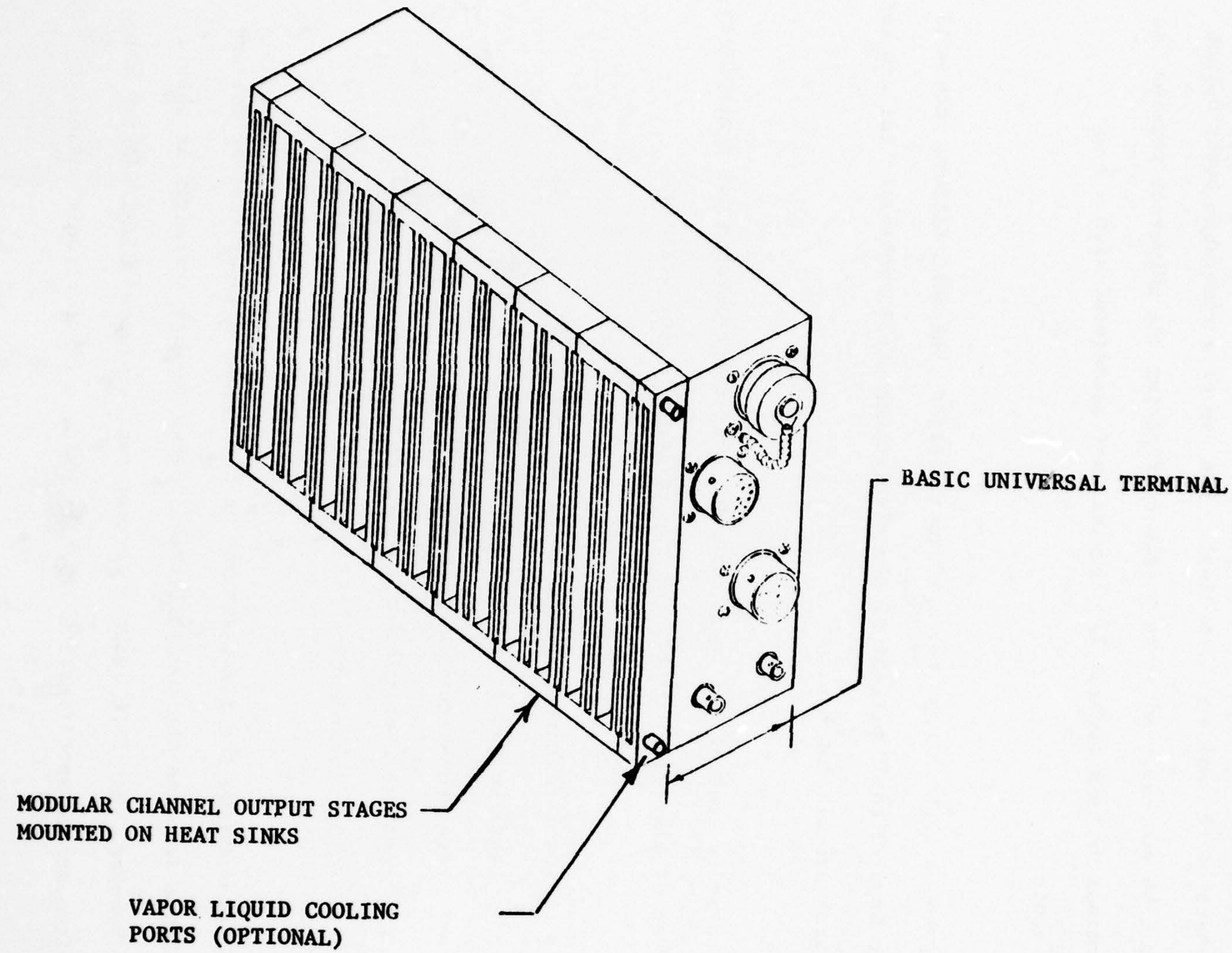


FIGURE 42 INTEGRATED LOAD MANAGEMENT CENTER CONCEPT

be provided as plug-in modular "strips" as illustrated in Figure 42 or as an integrated part of the universal terminal. The use of a removable power output section would be desirable so as to permit configuring the universal terminal as an ILMC terminal or as a general I/O terminal not associated with a load management center.

Whatever approach that is used for SSPC installation, the SSPC failure rate will be anywhere from 12 to 50 percent of the electromechanical controller failure rate of 20 failures per million hours.

Use of solid state controllers is recommended over electromechanical controllers for the advanced electrical system for the following major reasons:

- Lower failure rate (12 to 50 percent of EM)
- Lower weight and volume (25 to 27 percent of EM)
- Lower non-recurring cost (20 to 100 percent of EM)
- Lower turn-on/turn-off delays (5 to 16 percent of EM)
- Higher interrupting and overload capacity

These factors override the disadvantages of the slightly lower (2%) efficiency due to voltage drop and the possibility of higher leakage current. It should be noted, however, that SSPC leakage current can be dropped significantly below the 500 micro ampere specification limit by using a clamp circuit inside the SSPC.

Additional investigation of the feasibility of integrating the demultiplexer and load controller functions into a common WRA (Weapon Replaceable Unit) is

required. However, as presently envisioned that this LRU consists of a redundant multiplex/control section, a redundant power supply section, and a non-redundant power switch section. This LRU replaces the 64 LRU's presently used to implement the LMC function, i.e., the demultiplexer and 63 power controllers and the interconnect wiring. Functional block diagrams of the two concepts are shown in Figures 43 and 44. This new LRU, therefore, consolidates the control-logic functions presently being performed in each of these stand-alone WRA's. It is envisioned that microprocessor or programmable logic assembly (PLA) technology can be effectively used to accomplish the required functions and achieve the desired integration for producing an Integrated Load Management Center (ILMC).

Significant improvements that are yielded by the ILMC are:

- Reduced electronic complexity - approximately 50 percent of the aggregate electronics of 63 load controller electronics used for the signal level control functions and internal supplied power.
- Simplified signal interface - the elimination of the wire-terminations needed to interface the 63 controllers to the demultiplexer.
- Power reduction - reduction in the signal control power presently needed for EMI immunity which is of the order of 10 watts considering power supply inefficiency.
- Reduced size - consolidation of the 64 individual LRU's into the one LRU eliminates multiple housings, mountings, interconnect wiring and duplicate electronics.

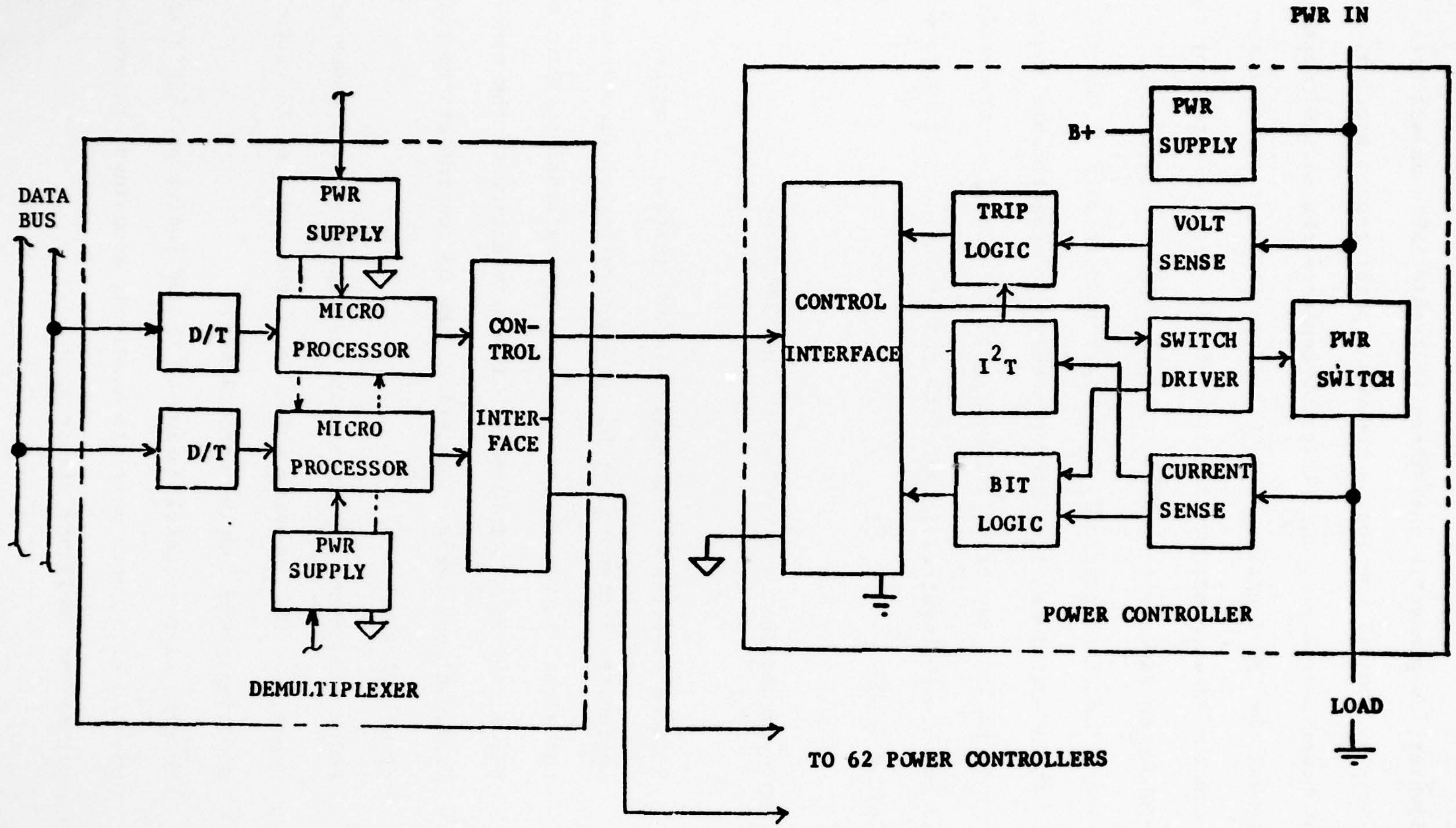


FIGURE 43 EXISTING LOAD MANAGEMENT CENTER CONCEPT

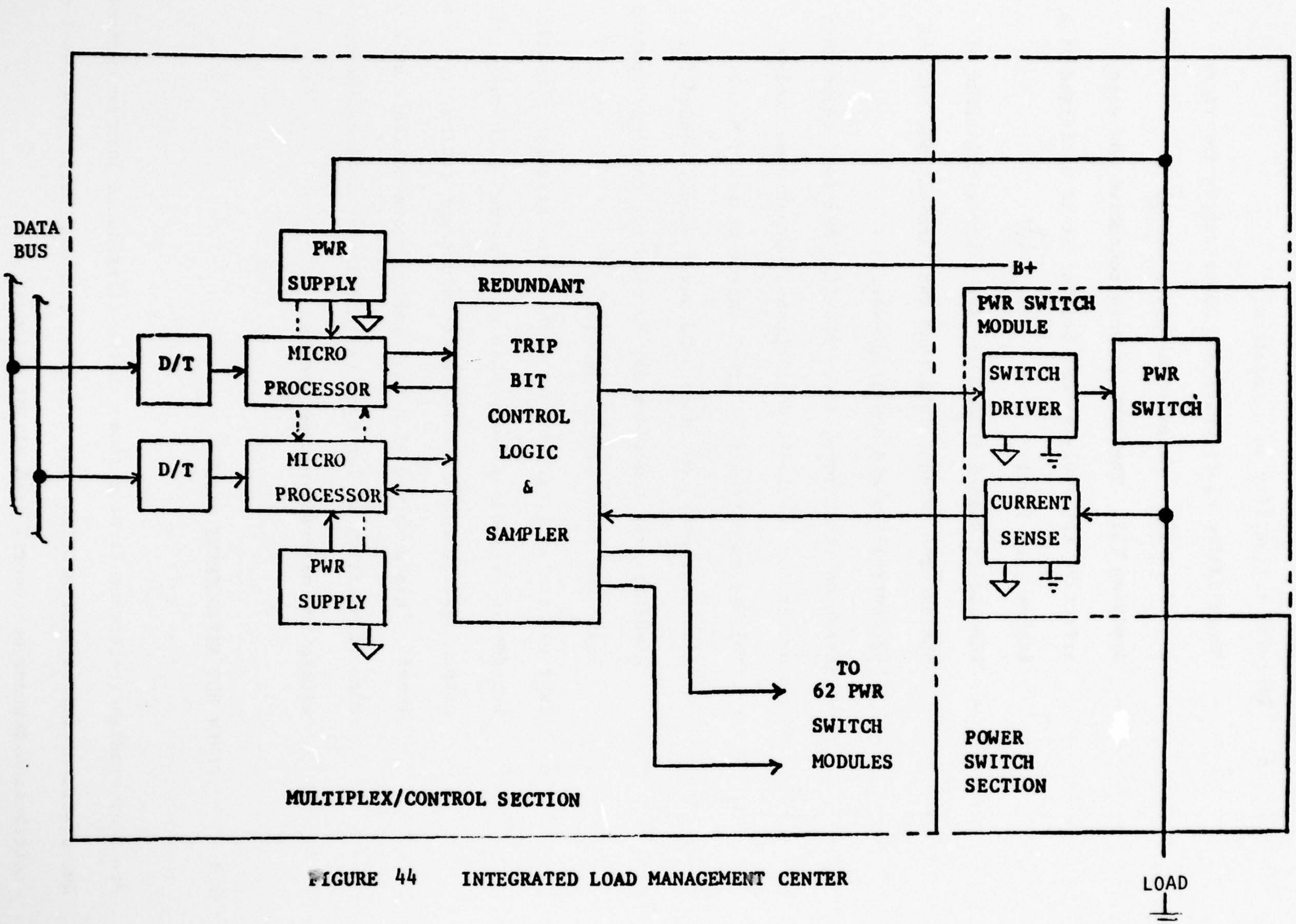


FIGURE 44 INTEGRATED LOAD MANAGEMENT CENTER

- Improved reliability as a result of:
  - Consolidated electronics and power supply functions significantly reduces total parts count.
  - Applying LSI. The ILMC is more conducive and cost effective for applying LSI because it is performed at a higher tier level.
  - Simplified signal interface, i.e., the elimination of the external signal interface and terminations between 63 controllers and demultiplexer.
  - Extended redundancy, i.e., the ILMC has full redundancy except for the individual channel output power stages. This is essentially a "free" improvement since the microprocessor controller and power supply needed for the demultiplexer functions may be used for the load controller logic and power supply functions.
- Improved system flexibility - perhaps the greatest potential benefit is derived by being able to "program" each individual output power channel for the desired current rating, i.e., trip level. This combined with the smaller, more reliable unit allows more versatility in locating the LMC in the optimum locations within the aircraft weapon system.

5.9 POWER BUS ARRANGEMENT STUDY

Previous studies (reference 2) have shown that a distributed load management center bus system minimizes system weight and decreases vulnerability. The level of bus distribution depends on several predominant factors:

- o Spatial distribution and installation density of utilization equipment
- o Total quantity of equipment, and
- o Packaging density limits for load controllers

The level of LMC distribution was studied in some detail in references 2 and 3 for the A-7D single engine attack aircraft. Results from these studies are summarized below.

Load management center locations in aircraft can be established by implementing one of two concepts. The first approach is to install one LMC at the aircraft load centroid. This LMC will service all aircraft loads and would therefore contain the full set of aircraft load controllers. In addition, this centralized LMC will contain the generation system point-of-regulation since no other LMC's exist. With this concept, the bus management system will be very simple since a sub-bus feeder network is not required. The power source feeders will route directly to the LMC rather than to a set of main power bus centers. The major problems with this concept are: (1) high level of vulnerability to battle damage, (2) large concentrated volume required for installation, and (3) total system weight is higher than with distributed LMC's due to the relatively long wires to individual loads.

The distributed LMC concept is the second and more reasonable approach. While this second concept requires establishment of main power bus centers and a sub-bus feeder network, the system vulnerability is significantly improved along with reducing the total weight of the bus management and

power distribution subsystems. Figure 45 depicts the results of a weight analysis to determine a "weight optimal" LMC distribution. As shown in the figure, the relative bus management subsystem weight increases as the number of distribution centers increase. The bus management weight includes feeders, feeder protectors and the load management center hardware. As the number of LMC's increase, the number of feeders and feeder protectors increase linearly. In contrast, the LMC weight change is non-linear due to packaging factors for the load controllers and associated hardware. As the number of load controllers in a given LMC decreases, the proportion of LMC housing and sub-bus hardware costs incurred by each controller increases.

While the bus management subsystem weight rises with an increase in LMC quantities, the power distribution harness weight decreases. The harness weight decrease follows the law of diminishing returns. After six distribution centers, the bus management weight increase exceeds the harness weight decrease for each additional LMC. As shown in the figure, however, a selected LMC quantity of between four and seven would have little impact on the system weight. The optimum number of LMC's for the A-7 example is therefore five. This representative bus layout is illustrated in Figure 1.

It should be noted that the bus management and power distribution weight for the distributed bus arrangement is approximately 80 percent of the centralized concept. Simply by changing the bus distribution concept, a 20 percent weight reduction can be achieved while at the same time, improving the system vulnerability. It should also be re-emphasized that the optimum LMC quantities will vary with aircraft specifically due to the distribution of utilization equipment in the aircraft.

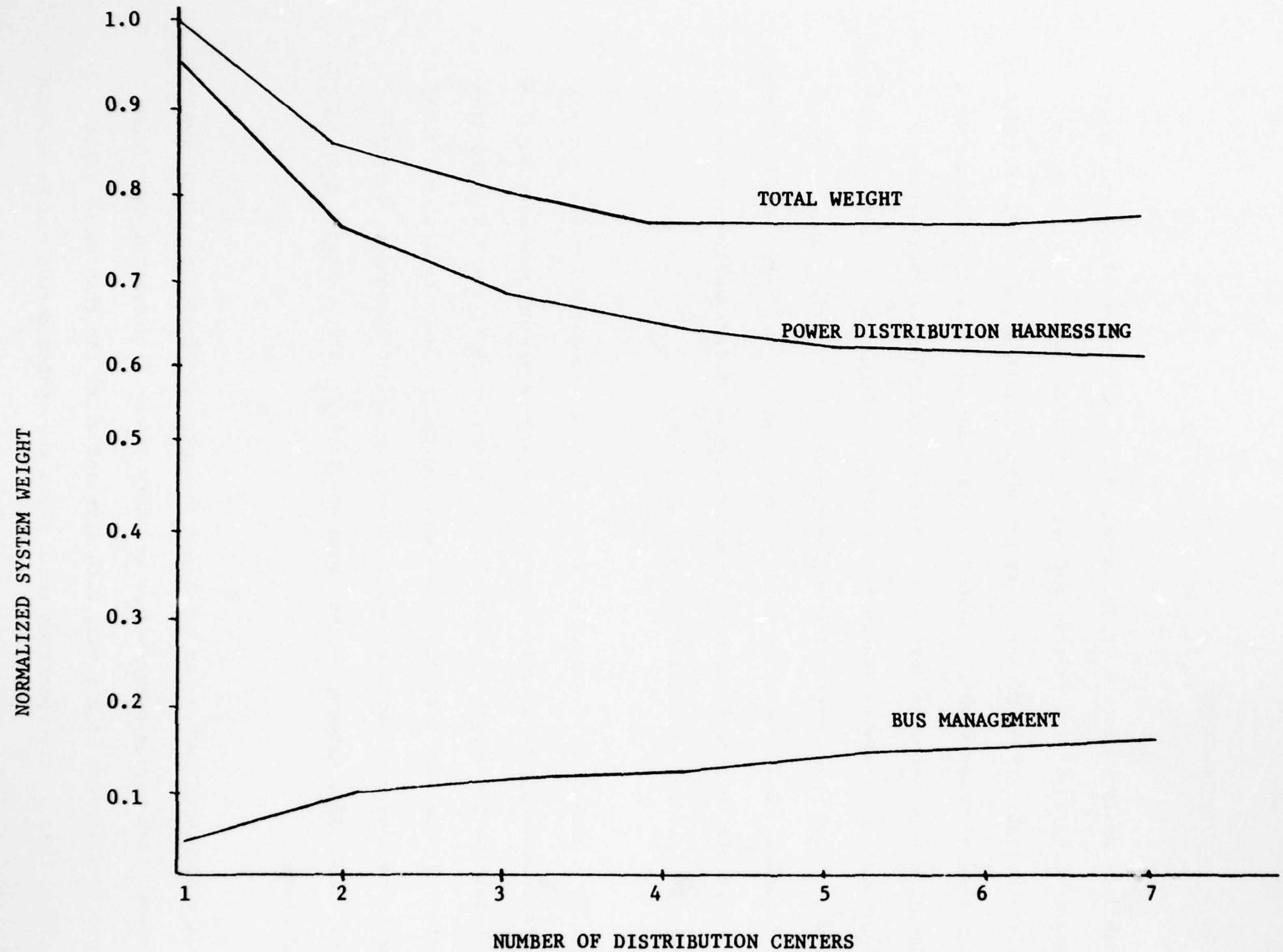


FIGURE 45 NORMALIZED SYSTEM WEIGHT VS DISTRIBUTION CENTERS FOR SINGLE ENGINE AIRCRAFT

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POWER SYSTEM CONTROL STUDY. PHASE I - INTEGRATED CONTROL TECHNI--ETC(U)

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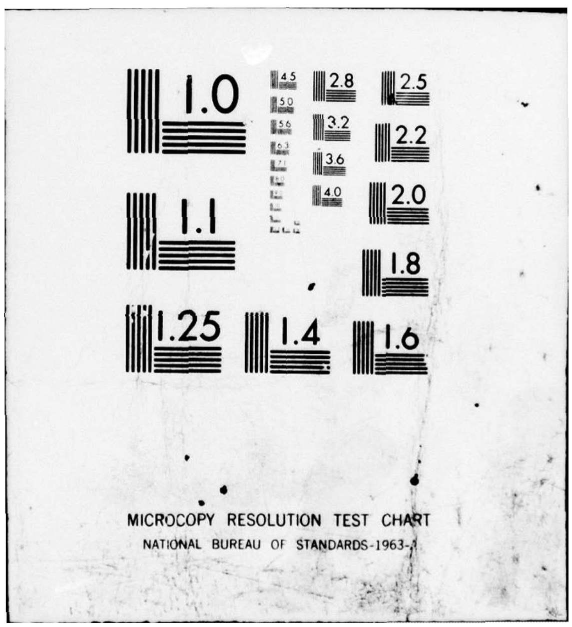

1.0
1.1
1.25
1.4
1.6
1.8
2.0
2.2
2.5
2.8
3.2
3.6
4.0
MICROCOPY RESOLUTION TEST CHART
NATIONAL BUREAU OF STANDARDS-1963-A

5.10 STABILITY ANALYSIS CONCEPTS AND PROCEDURES

5.10.1 INTRODUCTION

Stability analysis is the study, modeling and validation of control system dynamic behavior. Its primary purpose is to determine the stability or relative stability of a control system where stability is usually defined as that system property which causes the transient or frequency response to a disturbance to decay to zero or to some acceptably small oscillations. Alternately an unstable system is one which continues to oscillate or increase without bound upon application and/or removal of a bounded input or system disturbance. By these definitions, a stable system may still be considered unacceptable in that the decay (time constants) of the oscillations or other response features are outside of desired limits.

Usually a control system must be stable in order to serve a useful purpose wherein the study of dynamic behavior of a stable system becomes an analysis to determine the margin of stability and sensitivity of the system to disturbances occurring over its full operating range. Some typical performance specifications or figures of merit include transient overshoot, settling time, rise time, damping ratios, load sensitivity, phase margin, gain margin and others.

Highly developed techniques exist for the analysis of linear control systems - systems which may be described by ordinary linear differential equations whose coefficients do not vary with time and where the theorem of superposition holds. Superposition holds where the system output can be defined

as the sum of two or more outputs resulting from two or more inputs acting alone. The analysis of electrical power control systems is complicated due to the nonlinear characteristics of several of its components; namely, generator exciter and voltage regulator saturation, transformer/motor rectifier and other nonlinear load characteristics. Summarily, the analyses of electrical power control systems involve the analysis of nonlinear system where fortunately the nonlinearities often tend to make the system more stable.

A simplified block diagram of an aircraft electrical power system model is shown in figure 46. This is similar to the simple model analyzed in reference 9; however, the prime mover (engine), Constant Speed Drive (CSD) and EMUX load management loop have been added to this block diagram. The block diagram typifies the approach normally taken to dynamic analysis in that the next step is to formulate differential equations or state equations of each block relating input and output variables. These equations, even for a nonlinear system, can then be simultaneously solved numerically by computer and/or otherwise analyzed. Each block can constitute a system within itself. In figure 46, the blocks designating the engine and the CSD can be disregarded should constant speed (frequency) be assumed into the generator. The more detailed block diagram shown in figure 47 constitutes the Constant Speed Drive (CSD) and generator/regulator model used in reference 10.

The paragraphs to follow discuss typical aircraft power system transients and their affects on performance. Some tentative block modeling diagrams are established for the single-engine and multi-engine aircraft power system

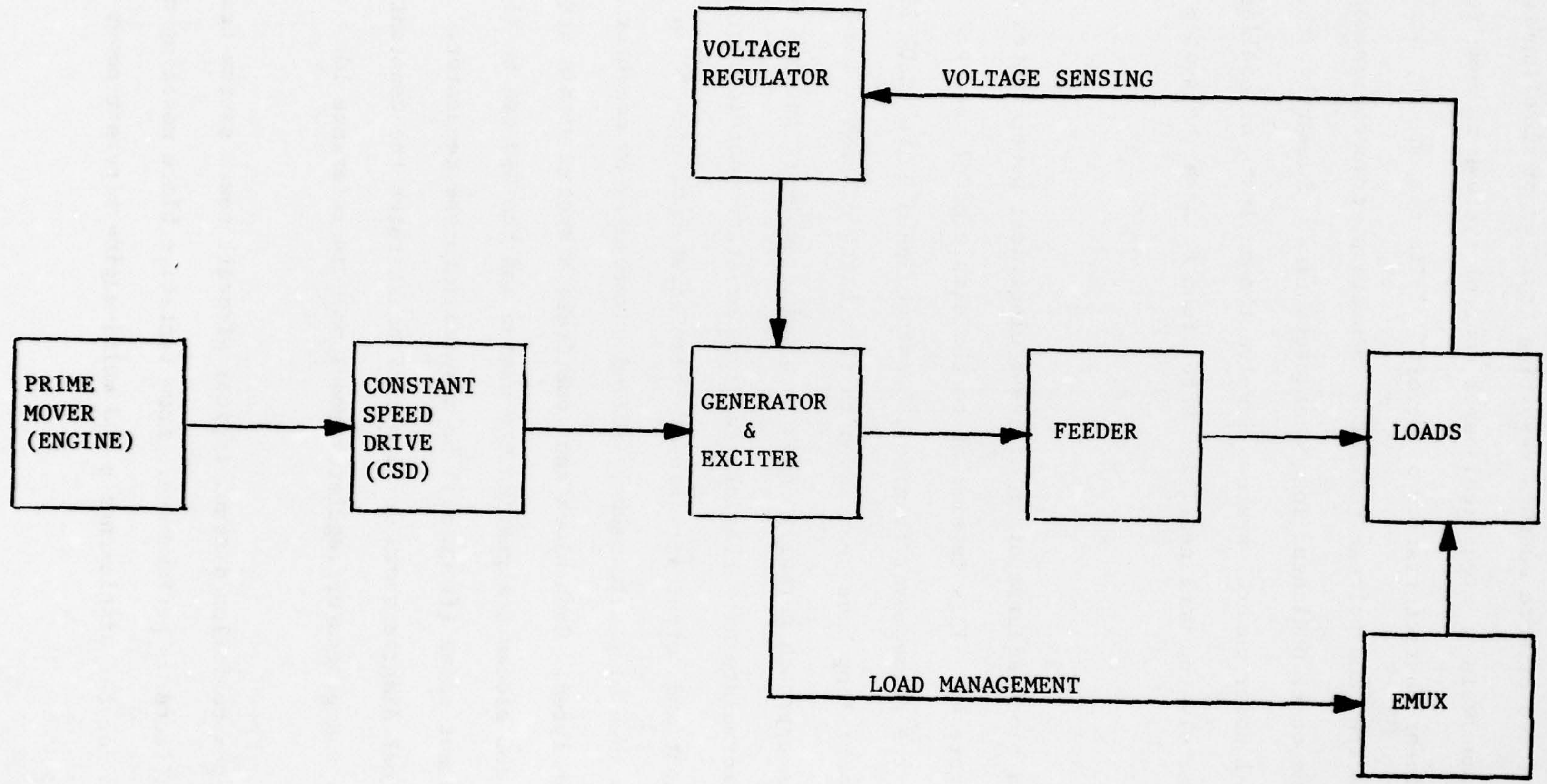


FIGURE 46. SIMPLIFIED BLOCK DIAGRAM - AIRCRAFT ELECTRICAL POWER SYSTEM MODEL

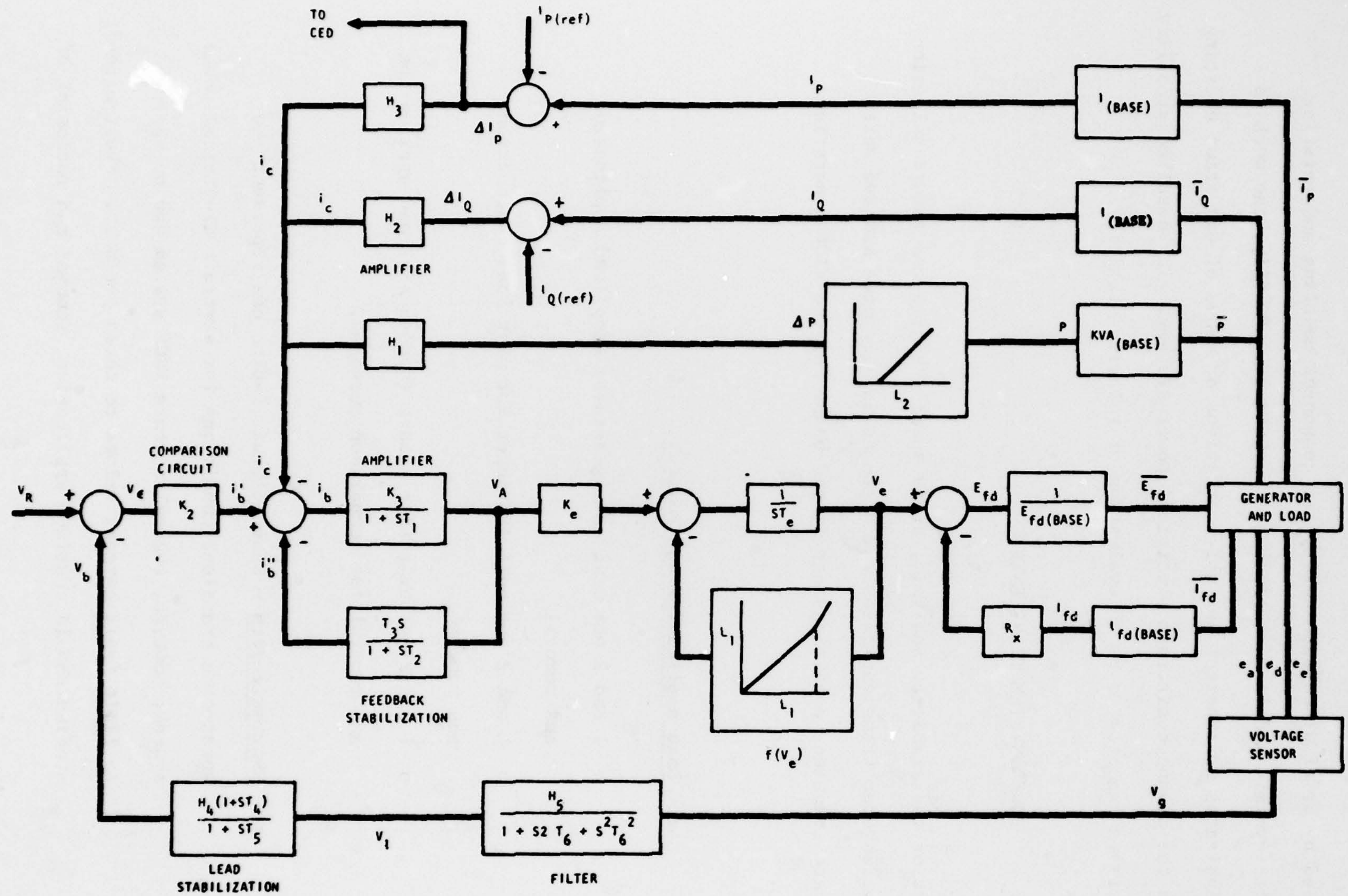


FIGURE 47 . CSD GENERATOR-REGULATOR BLOCK DIAGRAM

followed by brief discussions of system component modeling and existing computer dynamic modeling routines and techniques. Finally, the work to date indicates no inherent unstable conditions or modes of operation existing within the advanced aircraft electrical power systems. The detailed stability analysis is expected to be an evaluation of relative stability.

5.10.2 POWER SYSTEM TRANSIENTS

As part of the stability analysis, study should be directed toward selection of power system transient cases which are to be simulated and evaluated. Candidate transient conditions are listed below for the VSCF generating system:

a) Load Application and Removal

- 1 and 2 per unit main generator step load application and removal
- 1 and 2 per unit APU generator step load application and removal
- 1 and 2 per unit time phased (sequenced) load application and removal for comparison purposes.

b) Short-Circuits - Short-circuit faults can represent the worst case transient conditions for aircraft electrical power system; however, the solid state VSCF system can be expected to limit fault current values to three per unit. Short-circuit studies should consider application, removal and reclosing of

isolated and paralleled systems into short circuits located at generator terminals, converter terminals, and various bus levels. Types of short-circuits in usual descending order of severity for aircraft are listed below:

- o Single phase-to-ground
- o Double phase-to-ground
- o Balanced three phase

c) Paralleling - The paralleling of two AC power sources require close tolerances on phase angle and frequency between the sources at the time of paralleling. The minimum transient occurs where the phase angle and frequency are the same between the two sources, being the transient generated by redistribution of load between the two units. Candidate cases for momentary and/or long-term parallel operation are enumerated below:

- o Main generator with external power
- o Main generator with APU generator
- o APU generator with external power
- o Main generator with main generator (paralleled multi-engine system)

d) Failures - Candidate failure transients which should be considered are:

- o Sudden loss of generator excitation

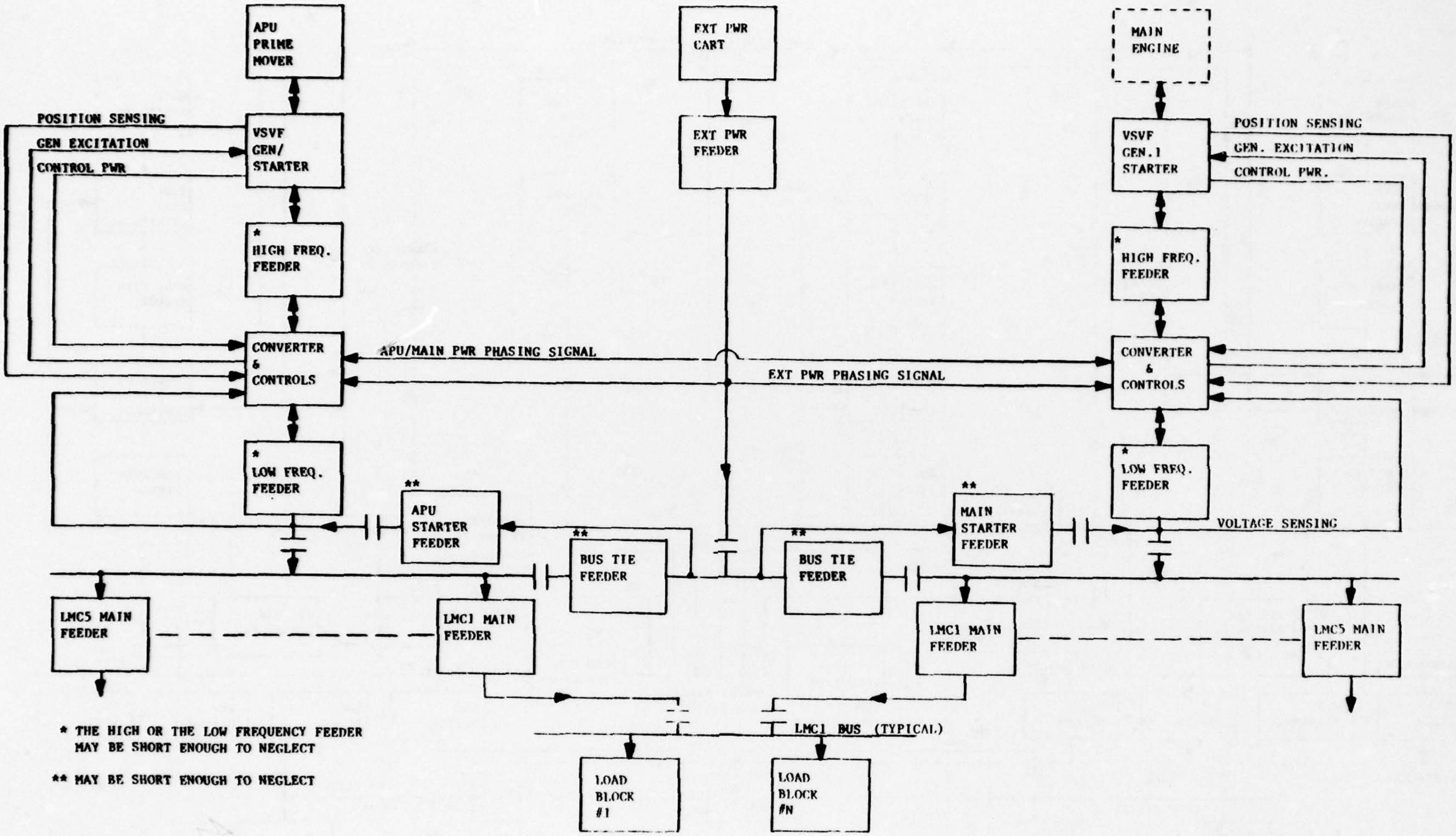
- Loss of phase voltage regulation
- Loss of real or reactive load division (parallel system)
- Open or shorted SCR in converter
- Misfiring of SCR in converter
- Open phase on generator or converter output feeder (parallel system)

e) Engine Starting - Engine starting modes of operation which can have significant transient impact are enumerated:

- Main engine generator powering main engine starter/generator
- APU generator powering main engine starter/generator and conversely
- External power supplying main engine starter/generator
- External power supplying APU starter/generator (External power sources are considered beyond the scope of work, but must be mentioned for completeness.)

5.10.3 BLOCK DIAGRAMS AND ELEMENT MODELING

Tentatively detailed system block diagrams for modeling the advanced single-engine and multi-engine power system concepts are depicted in figures 48 and 49. These diagrams intentionally show detailed feeder and other elements some of which may be neglected or combined once applied to a specific design. For example, should the converter be located at the generator, the impedance



* THE HIGH OR THE LOW FREQUENCY FEEDER MAY BE SHORT ENOUGH TO NEGLECT

** MAY BE SHORT ENOUGH TO NEGLECT

FIGURE 48. BLOCK DIAGRAM - STABILITY ANALYSIS FOR ADVANCED SINGLE-ENGINE AIRCRAFT POWER SYSTEM

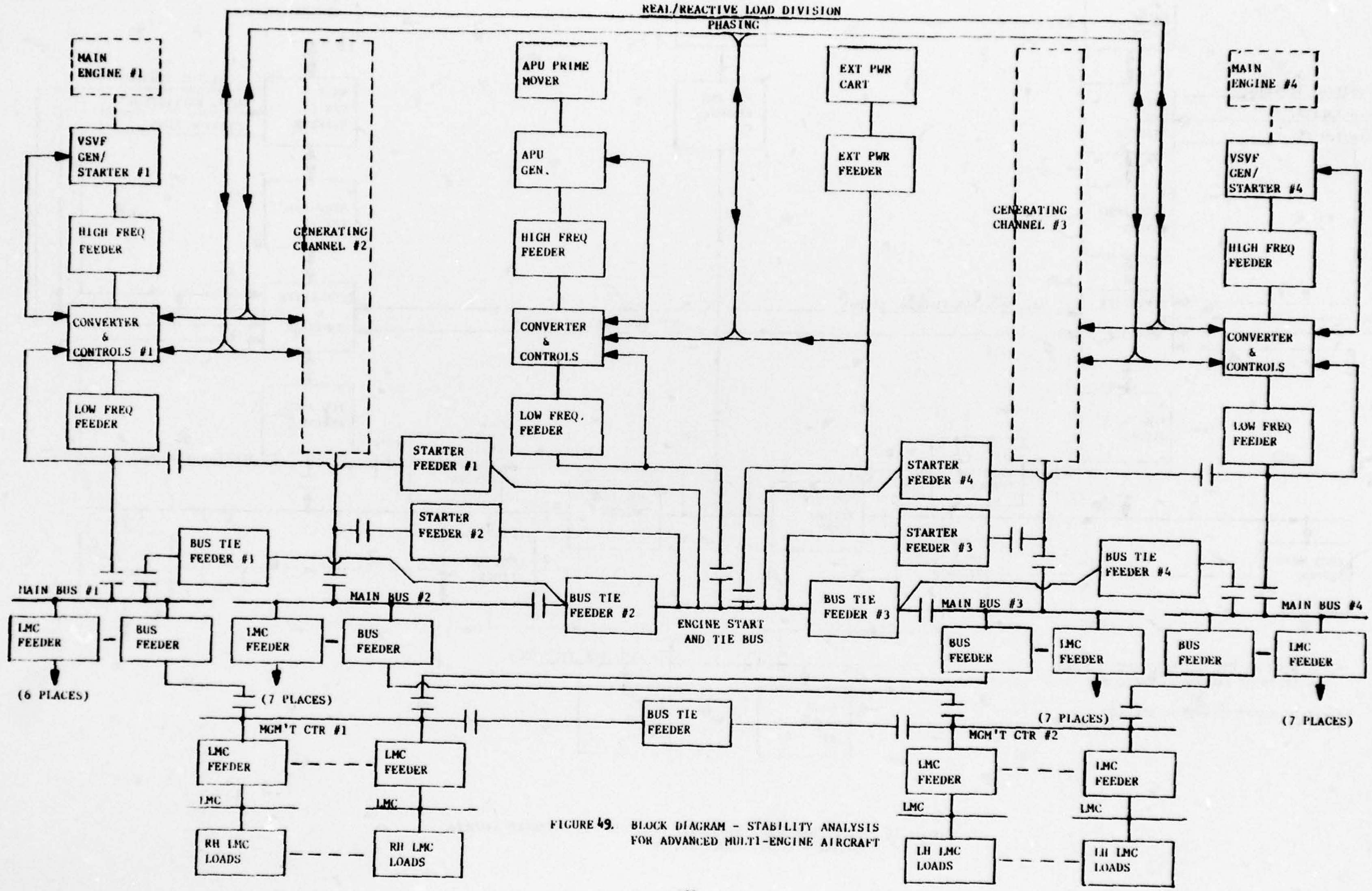


FIGURE 49. BLOCK DIAGRAM - STABILITY ANALYSIS FOR ADVANCED MULTI-ENGINE AIRCRAFT

of the high frequency feeder could be neglected; conversely, should the converter be located at the bus, the low-frequency feeder impedance could be neglected. Summarily, the elements to be modeled consist of the prime mover, the generator/exciter/regulator/converter section, power feeders and loads. Simulation of power contactors and solid state power controllers can be incorporated into bus and load feeders elements.

Modeling of prime movers must consider the dynamics of main engines, auxiliary power unit drives, and external power cart drives. Main engines can be considered as stiff sources while driving main generators; however, the analyst may need to evaluate affects of engine speed variations over a tupical segment of the flight profile. Again, in the interest of model simplification, engine speed variations may be neglected where effects are determined to be minimal.

The generator, exciter, voltage regulator/converter can be modeled as an integrated subsystem or as separate control elements. Considerable work has been expended in modeling commercial and utility type power generation systems (references 12 through 16). The VSCF generator/converter system has been modeled earlier and is described in detail in references 10 and in appendix B. However, there are several modifications, such as zero sequence feedback, to be made to these models in order to update them to the latest configuration for the Phase II stability analysis efforts.

The generator, load center and bus tie feeders are to be included in the stability analysis modeling to account for the damping effects of resistance and the energy storage and transient effects of feeder inductance. The

geometry of phase feeder installations and the skin, proximity and ground plane effects on feeders must be included in establishing feeder parameters. With the exception of larger loads, the load feeders may possibly be neglected.

Electrical load modeling for commercial electrical power utility systems has received some attention in recent years (reference 11). Appendix C contains the abstract to the work performed in this reference. Some attention was devoted to aircraft load modeling in reference 10. However, the aircraft power system analyst should perform the following activities in regard to load modeling:

- o Survey and catalog aircraft load types and the general location of each type.
- o Define load types and characteristics with particular emphasis on the dynamic behavior and nonlinear characteristics of loads such as motors, transformers and electronic equipment power supplies.
- o Perform preliminary evaluation of load types and their effects upon overall power system dynamic behavior, determining the detail level necessary for load modeling.
- o Select appropriate load models, probably modeling blocks of load types on a Load Management Center bus basis rather than individual loads.

5.10.4 PROGRAM DEVELOPMENT AND VALIDATION

Most system's dynamic and stability analysis are presently performed by digital computer simulation. The system is modeled by a set of element control blocks, each block having its inputs and outputs defined by time varying differential equations which are usually reduced to the state variable form. Thus, the system solution is basically the solution of a set of simultaneous differential equations with a given set of initial conditions. Elements with nonlinear characteristics are defined by tabular inputs. The solution to the usually nonlinear equations are performed by numerical methods.

There are several existing digital computer dynamic simulation programs. Typical existing programs are listed below.

a) GTSAR, Vought Corporation's segmented computer routine performs simulation, quasi-linearization and stability analysis for a mathematically modeled multi-ordered system.

b) EASY Dynamic Analysis Program (as applied to Aircraft Electrical Power Systems Modeling), Boeing Aircraft Company's detailed model as developed for the Air Force. The program generates the model and performs various types of linear and nonlinear simulation analysis.

c) DYSIM, General Electric's time-shared dynamic system simulation library routine.

The power system analyst should review these and other applicable routines to determine their suitability for performing the stability analysis. Often the generality of these routines force compromises in the element models. Should the routines prove to be inadequate or do not properly model the system elements, then new programs must be developed or modification made to existing routines. A simplifying assumption most often made is that the power system is balanced, i.e., balanced load or balanced short-circuit (three phase fault). These assumptions are good for commercial systems but a phase ground fault on aircraft gives the worst case short-circuit current. Thus, the power system analyst must decide whether to model the system as balanced or unbalanced. The unbalanced model being an order of magnitude more complex (considering the three phases and cross coupling between phases). However, the VSCF incorporates fault current limiting perhaps allowing the analyst to treat it as a balanced system.

The power system analyst has a most difficult task in defining the model type and level of detail as well as formulating the system equations. Once the mathematical model has been established and run, another most important task is to validate the computer model with actual measured system responses. These actual responses for comparison may be obtained from system operations, system qualification test data or tests from similar systems.

5.11 SAFETY DESIGN CRITERIA AND REQUIREMENTS

Safety design criteria was a consideration in the evolvement of the electrical system as implemented using advanced integrated control techniques. In most conventional aircraft, primary flight controls are mechanically actuated, hydraulically powered with adequate aerodynamic stability to permit all flight phases without electrical power. Criteria applicable to these older or less advanced aircraft basically remains applicable to aircraft employing the advanced integrated control techniques described in this report. The task of compiling a safety design criteria for aircraft advanced electrical systems is influenced by several factors. Until the advent of computer controlled airborne electronic equipment in manned aircraft, electrical system design has been relatively safe. The problems which have arisen have not been associated with difficulties in the operating of the electrical system, but primarily with difficulties in fabrication and installation of that system. Design criteria for aircraft electrical power systems have addressed, for the most part, practices which attempt to reduce or eliminate hazards caused by placing the electrical system within the airframe.

Traditionally, the number of aircraft accidents directly caused by electrical system failures has been much smaller than that caused by other aircraft systems, such as hydraulic and propulsion. There are two means by which an electrical system can threaten the safety of the aircraft. Either the power system fails resulting in loss of some critical function (e.g., the all attitude instrument system) or the electrical system causes a fire

(e.g., wire bundles chaffing against aircraft structure or high pressure hydraulic hoses). Prior efforts that have been expended to insure that these types of failures do not occur apply equally to aircraft using advanced integrated technologies.

In some respects, these problems may be less likely to occur with a solid state system. There are fewer wire bundles, and those that exist are substantially reduced in volume which allows the system to be more ideally installed (location plus protection). There are also fewer connectors and the power controlling units are self contained.

In the future, however, the advent of control by electronic signals commanded by computers will require complex redundancy which implies a growth over present conventional wiring and connectors. The chances for a failure which could lead to loss of power or to fire substantially increases. In addition, the results of power loss become catastrophic. To reduce these hazards, solid state technology should be used. Multiplexing reduces the number of places that wiring failures can occur. Most importantly, the EMUX system can be made to meet the redundancy requirements without complicating and enlarging the overall electrical system. This requirement will become paramount in FBW (Fly-By-Wire) aircraft.

The advanced integrated electrical system offers operational advantages not possible in conventional electrical systems. The advantages are attributed to the BIT and load monitoring functions contained in the EMUX system. In order to take fullest advantage of these capabilities, operational flight phases can be identified and load/equipment priorities established.

5.11.1 SAFETY DESIGN REQUIREMENTS FOR THE ADVANCED AIRCRAFT ELECTRICAL SYSTEMS

The following is a list of safety design requirements applicable to all aircraft electrical systems including systems containing integrated control techniques.

1. Route wires and locate components so they do not create interference with adjacent systems.
2. Design systems with an absolute minimum of connections and terminations.
3. Ensure that primary and redundant system circuits are not supplied from the same power bus or power controller.
4. Ensure that elements of a redundant system do not pass through the same connector used by elements of the primary system.
5. Avoid termination of power and signal leads on adjacent pins of connectors.
6. Provide supports to prevent abrasion or chafing of wires and cables.
7. Route wires that are attached to normally moving parts to twist with rather than bend across adjacent moving parts.
8. Do not allow several critical electrical components to be protected by a single power controller.
9. Provide electrical shielding wherever it is necessary to suppress radiofrequency interference and other sources of spurious energy.

10. Provide protective devices for primary circuits and equipment to protect from overloads.
11. Ensure that system operation is not degraded by temperature extremes.
12. Avoid routine electrical wire bundles adjacent to fuel lines, hot air ducts, or mechanical linkage.
13. Locate wire bundles away from ejection controls.
14. Design to enhance protection from ligntning strikes.
15. Provide positive protection for terminal blocks to prevent short circuits resulting from contact with debris or elements of the environment.
16. Main power ON-OFF switch located on the equipment is clearly labeled (where applicable).
17. All equipment external parts, surfaces and shields (exclusive of antenna and transmission line terminals) are at ground potential at all times during normal operation.
18. External or interconnecting cables must have a ground wire in the cable terminated at both ends in the same manner as the other conductors, (ground is part of the circuit).
19. Except for coaxial cables, shields are not used for current-carrying ground connections.
20. The path to ground from equipment:
    a) Is continuous
    b) Has ample carrying capacity to conduct any operating or fault current imposed.

c) Has impedance low enough to limit the potential above ground and facilitate the operation of overcurrent devices in the circuit.

d) Has inactive wires grounded that are installed in long lines (conduit or cables).

e) Has sufficient mechanical strength of the material to minimize possibility of ground disconnection.

21. Shielding on wire or cable is secured to prevent contact with exposed current-carrying parts or grounding to the chassis at any point other than the ground termination.

22. Shielding on wire or cable ends at sufficient distance from exposed conductors to prevent shorting or arcing.

23. Connectors providing separation of, or connection to, multiple electric circuits are designed such that it is impossible to insert the wrong plug in a receptacle or other mating unit.

24. Where the above item is not practical, mating plugs and receptacles are clearly coded or marked to indicate mating connections.

25. All remotely located assemblies have provisions for safety switches to allow independent disconnect in the associated equipment.

26. Materials, as installed in the equipment and under service conditions specified in the specific equipment specification, do not liberate gases which combine with the atmosphere to form an acid or corrosive alkali, nor liberate toxic or corrosive fumes which would be detrimental to the performance of the equipment or health of equipment operators.

27. Failure of one phase of a three-phase power electrical system does not result in an unsafe condition.

28. For those equipments which require both ac and dc, power failure of either dc or ac power source does not result in an unsafe condition.

29. Dissimilar metals are not used in intimate contact within electrical systems unless suitably protected against electrolytic corrosion.

30. Static ground provisions are made for the discharge of accumulated charges of static electricity by bringing the aircraft to ground potential.

31. Electrical equipment is installed with considerations to protection of exposed terminals by orientation or insulating covers and is located below drip points and tube fittings carrying fluids.

32. The generator is capable of operation at 110 percent of the maximum rated input speed for 5 minutes after having stabilized thermally and while operating at full rated load and minimum input speed.

33. Under any normal system operating conditions including initial power up and application and removal of loads, overloads, and faults; nuisance tripping does not occur. Protection from the following malfunctions is provided: overvoltage, undervoltage, under frequency, feeder fault, extraneous frequency content, symmetrical component voltage content, generator underspeed.

34. Electrical connections are so arranged and wired that hot leads are not terminated in pins or other exposed contacts which might be accidentally shorted or touched.

35. All electrical components are made explosion proof; i.e., so that units or components cannot cause ignition in an ambient explosive gaseous mixture with air.

36. Safety margins for equipment are used if EMC problems may result in catastrophic failures. Unless otherwise specified, safety margins less than 6 db (20 db for explosives) are not used.

37. Wiring and cabling are designed to minimize coupling and obtain optimum separation and use of available wiring space. Cable design includes provisions for adequate termination of shielded wires.

38. The system design includes provisions for protection of personnel from R-F hazards, electromagnetic, electro-static, and shock hazards in accordance with the requirements in MIL-STD-454. When protection by design is not technically feasible, adequate safety precautions are included in operating and maintenance manuals.

39. Electrical bonding connections are so installed that vibrations, expansion or contraction, or relative movement incident to normal service use will not break the bonding connections nor loosen them so that the resistance will vary during movement.

40. Bus switching schemes does not allow total electrical failure to occur due to a single point fault within the switching or bus control circuits.

41. Interlocks required to control critical subsystem operation are capable of being functionally checked prior to flight, or if that is not practical, the continuity of the control system have provisions for checkout prior to flight.

5.11.2 SURVIVABILITY/VULNERABILITY

Survivability/vulnerability criteria were also considered in the evolvement/ application of advanced integrated control techniques. The following paragraphs provide the survivability/vulnerability considerations relevant to comparing conventional electrical systems with advanced control techniques.

Survivability of aircraft electrical/electronic systems can typically be improved by the following methods:

- o Vulnerable areas reduced by using smaller components and relocated to less vulnerable locations.
- o Existing nonredundant systems are replaced with redundant systems. A "black box" which is internally duplicated is not regarded as redundant in terms of survivability. Redundant components must be physically separated so that any one combat hit cannot destroy both components.

The advanced control techniques enhance survivability through reduction of vulnerable areas (especially wire bundles) and through reduction of single-point failure sites by incorporating redundant systems. A review of the design shows the following:

- The multiplex terminals, as well as the processor, are redundant. As a minimum, redundancy is used for system elements which would cause a mission abort. This includes adequate physical separation as part of the redundancy. Placement of these items are typically governed by space available, but should follow a rule of thumb on placement of components on opposite sides of the aircraft. Use of existing aircraft structure for shielding should also be employed as practicable.

- The multiplexer data transmission line should be redundant and physically separated. It is noted that a single data bus does not fulfill this requirement, therefore, the redundant data bus used in the recommended design meets the survivability guidelines.

- In general, the control wire (signal source to multiplexer) should be redundant for mission critical items in the same manner as the transmission line, however, this requirement will not apply if the pair of wires is short and routed to utilize existing protection (cockpit armor, aircraft structure, etc.).

- In the power switching subsystem, the load management centers should be physically protected, wire runs made short, and load management centers be isolated.
- For flight critical systems, e.g., fly-by-wire, completely redundant power channels should be provided.

SECTION VI
CONCLUSIONS

The Phase I objectives have been concluded as reported herein. The conclusions are as follows:

6.1 POWER GENERATION

The power generation requirements for the 1990 time period can be met with both the VSCF (cycloconverter) and the IDG concepts. Definition and evaluation of specific weapon system mission and performance requirements will dictate which of the two concepts is optimum. The CSD (drive separate from generator) is not considered a viable system for new electrical system designs, primarily because of the weight penalty imposed by this system.

6.2 ELECTRIC ENGINE START

Electric engine start can be provided by both the VSCF and IDG technologies. Significant advantages for multi-engine aircraft and marginal advantages for single engine aircraft are achieved over conventional self-start concepts under the following conditions:

1. An APU driven generator is provided with rating sufficient to start the engine.
2. A generator is provided on each engine.

3. The rating of the engine mounted generator is established by the utilization equipment load demand plus growth requirement and is sufficient to start the engine.

6.3 EMUX INTEGRATED GENERATOR CONTROL

It is not feasible to perform the GCU functions (regulation and protection) within the EMUX system because of the long throughput time required by EMUX. GCU interface with EMUX for load management and BIT functions is feasible. Implementation of the GCU is quasi-feasible in performing BIT (Built-in Test) functions and some control functions. Because of response requirements and signal conditioning complexity, it is not recommended that the generator regulation and protection functions be performed with the microprocessor.

6.4 "GAPLESS" POWER

A true "gapless" power bus covering all contingencies is not possible in an AC system. Power interruptions less than 20 milliseconds are possible by establishing a bus of limited power capacity and using solid state or hybrid controllers for bus transfer.

6.5 AC BUS CONTROLLERS

Electromechanical contactors are best for high current bus (line and bus-tie) controller applications. Solid state and hybrid controllers can perform switching functions not possible with electromechanical devices and are recommended for use for low current and other specific applications requiring improved switching speed and overall improved switching performance.

6.6 POWER DISTRIBUTION AND CONTROL

An EMUX implemented power distribution system offers several advantages over the conventionally implemented system. The EMUX system includes signal input stimulus with signal sources, power switching and circuit protection with power controllers, and computerized control logic implementation. Application of EMUX is recommended for all military aircraft of moderate to high electrical/electronic complexities. Application on low complexity type aircraft may also be desirable or dictated where emphasis is placed on performance and application flexibility or where further development (use of more MSI/LSI to accomplish circuit functions) of EMUX is pursued.

SECTION VII

RECOMMENDATIONS

It is recommended that the planned work for Phase II of the program should proceed. It is further recommended that the system modeling and stability analysis to be conducted during Phase II be done on the VSCF generating system.

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APPENDIX A

IDG RELIABILITY AND MAINTAINABILITY

SUPPORT DATA

ADVANCED TECHNOLOGY 90 KVA SYSTEM

ADVANCED TECHNOLOGY 90 KVA SYSTEM

INTEGRATED DRIVE GENERATOR

The engine mounted IDG with side by side integral generator converts variable speed shaft power to 400 Hz electrical power. The configuration represents an evolutionary improvement in packaging of proven spray-oil cooled generator and speed conversion hardware into a single housing.

Specifications

Rating (per channel)

Continuous	- 90 KVA
5 Minutes	- 112.5 KVA
5 Seconds	- 150 KVA
Power Factor	- 0.75 lagging to 1.0
Output Frequency	- 400 Hz
Input Speed	- 4500 - 9075 rpm
Output Voltage	- 115/200 VAC (3 phase)

GENERATOR CONTROL UNIT

The associated GCU contains all the circuitry necessary to regulate the voltage, and control and protect generating channel. In addition, built-in-test equipment (BITE) circuitry is included to provide on-aircraft fault isolation.

BUS PROTECTION UNIT

The BPU contains the circuitry necessary for external power monitoring and protection, control of load shedding, tie bus differential breakers, and various ground service relays. When required, one BPU automatically supervises the electrical system whether it is single or multi-channel and ensures proper and safe selection and sequencing of breaker/contactor operations. Microprocessor based circuitry forms the basis for the BPU design. All of the system troubleshooting information that the BITE generates can be sequenced onto the alphanumeric display of the BCU.

CURRENT TRANSFORMER ASSEMBLY

The CTA comprises three toroidal transformers in a single package. One generator phase lead passes through each transformer, providing a single turn primary. The secondary windings supply an output current to the GCU and BPCU.

AUXILIARY GENERATOR

For systems requiring an APU of equal rating to the main channel, the APU generator is identical to the IDG generators except that it is contained in its own housing and is mounted on the aircraft APU. The spray-oil cooled generator electromagnetic components are interchangeable with those in the IDG.

MAINTAINABILITY AND RELIABILITY

The IDG is designed for "on condition" operation with only periodic inspection of oil level and filter required. The single-split line and modular construction of subassemblies allows for quick, easy replacement of all internal components. Shop maintenance of the controls is enhanced by automatic troubleshooting tests and information display.

The simplicity of concept, the conservatism in design and a significant reduction in parts result in enhanced reliability over previous designs. The high reliability of the controls is assured by the use of "hi-rel" components and by operating these components at less than rated values.

BITE (BUILT-IN-TEST EQUIPMENT)

The BITE system is a microprocessor based monitor which senses system information flow and detects failure symptoms without the need for additional aircraft circuitry. The microprocessor is programmed to allow the BITE to sequentially check the significant system characteristics over 50 times a second. From this information, the system status is determined relative to preprogrammed limits. When the BITE detects a failure symptom, including intermittent interruptions, a series of tests will isolate the fault and store the information in a nonvolatile memory. The memory retains data for on-aircraft troubleshooting and for subsequent maintenance shop readout.

APPENDIX B

EARLY VSCF MATHEMATICAL MODEL

VSCF MATHEMATICAL MODEL

Mathematical models are presented for three VSCF systems of the cycloconverter type. 60 and 90 KVA systems at 115/200 volts are presented and a 150 KVA system at 230/400 volts.

The 60 KVA VSCF is assumed to have a 6 phase generator. The 90 and 150 KVA systems are assumed to have 9 phase generators. Increasing the generator phases saves filter weight and lowers the per unit source impedance at the cost of increased numbers of SCR's.

Cycloconverters have a limited ability to handle reactive power. An override circuit which depresses the system voltage when the reactive load limit is exceeded is assumed. The system would normally be designed to supply 2 PU @ .75 power factor at full voltage and reactive power greater than included in this load would cause voltage droop. The VSCF will also have an absolute current limit which would be adjusted from 2 to 3 PU depending on the fault current requirements. A short circuit on the system does not appear as a short circuit to the generator, only as a large current low power factor load because the

converter SCR's are phased back to reduce the output voltage to whatever is required to drive the specified fault current.

Parallel operation is assumed to be in a phase locked mode in which one system is chosen as master and all other systems are slaved to it with phase lock loops. If accurate load division is required, the phase lock may be trimmed by a slow limited output load division circuit.

The source impedance of a cycloconverter has both resistive and reactive components with the resistive part tending to be largest. Voltage adjustment for load division should be proportional to the component of voltage drop in the source impedance caused by circulating power which is in phase with the terminal voltage. Phase or frequency adjustment should be sensitive to the source drop at right angles to the terminal voltage. Voltage adjustment for parallel cycloconverters tends to be largely done by real circulating power and frequency or phase adjustment by reactive power which is the reverse of conventional practice with synchronous machines whose source impedance is largely inductive.

Block 1 - Filter - Series RL - Shunt C

R is the resistance of the interphase transformer and converter wiring plus the equivalent commutating resistance $PL'f/g$ where

- P is the number of generator phases
- L' is the machine subtransient plus transmission line inductance
- f is the machine frequency
- g is the number of rectifier groups per bank

A six phase rectifier with a two leg interphase transformer has two groups of three rectifiers with 120^{o} conduction. A six phase rectifier could also have a three leg interphase transformer with three groups of two rectifiers.

having 180° conduction.

R also includes the machine and transmission line resistance per phase divided by g.

L is the interphase transformer leakage inductance plus the sum of the machine subtransient and transmission line inductance divided by g.

C is the filter capacitance.

	60 KVA	90 KVA	150 KVA
R Ohms	.11	.05	.12
L Microhenry	37×10^{-6}	20×10^{-6}	49×10^{-6}
C Microfarad	310×10^{-6}	350×10^{-6}	130×10^{-6}
Transfer Function	$\dfrac{1}{\dfrac{S^2}{9300^2} + \dfrac{2(.15)S}{9300} + 1}$	$\dfrac{1}{\dfrac{S^2}{11900^2} + \dfrac{2(.1)S}{11900} + 1}$	$\dfrac{1}{\dfrac{S^2}{12500^2} + \dfrac{2(.1)S}{12500} + 1}$

Block 2 - Converter

Error signal to output voltage gain is determined by the ratio of output voltage with full modulation to firing wave voltage.

The delay is one half the period between SCR firings. The values given are at a midspeed generator frequency of 1670 Hz.

Transfer function	60 KVA	$20\,e^{-30S} \times 10^{-6}$	Volts/Volt
	90 KVA	$20\,e^{-20S} \times 10^{-6}$	
	150 KVA	$40\,e^{-20S} \times 10^{-6}$	

Block 3 - Error Amplifier

Simple gain with negligible phase shift.

Transfer function $\quad 3$ Volts/Volt

Block 4 - AC Feedback

Used to improve transient performance.

Transfer function

60, 90 KVA $\quad \dfrac{.012}{1 + .00013S} \quad \dfrac{\text{Volt}}{\text{Volt}}$

150 KVA $\quad \dfrac{.006}{1 + .00013S} \quad \dfrac{\text{Volt}}{\text{Volt}}$

Block 5 - DC Feedback

Used to suppress any DC voltage in output.

Transfer function

60, 90 KVA $\quad \dfrac{1.67}{(1. + .016S)\,(1 + .001S)}$ Volts/Volt

150 KVA $\quad \dfrac{.83}{(1 + .016S)\,(1 + .001S)}$

Block 6 - Waveshaping Feedback

Used to minimize low harmonics of 400 Hz.

Transfer function

60, 90 KVA $\quad .0064\,\dfrac{25 \times 10^{-8}S^2 + 4.2 \times 10^{-5}S + 1}{4.4 \times 10^{-8}S^2 + 4.5 \times 10^{-4}S + 1}\,\dfrac{V}{V}$

150 KVA $\quad .0032$ ________________

Block 7 - Voltage Regulator Sensing

Transfer function 60, 90 KVA $\frac{.33}{1 + .001S}$ $\frac{\text{Volt}}{\text{Volt}}$

150 KVA $\frac{.17}{1 + .001S}$ $\frac{\text{V}}{\text{V}}$

Saturated output of this block is 12 volts for phase A, 11.7 volts for phase B, and 11.4 volts for phase C.

Block 8 - Regulator Gain

Transfer function $\frac{100}{1 + 4.7S}$ $\frac{\text{V}}{\text{V}}$

Block 9 - Wave Generator

Puts out 400 Hz reference whose amplitude is varied by DC voltage level of block 8 and whose frequency may be adjusted by block 14 when in parallel.

Transfer function Block 8 input $\frac{1.75}{(1 + .00004S)^2}$ Volts/Volts

Block 14 input 4 $\frac{\text{radians}}{\text{sec}} / \text{Volt}$

Block 10 - Reactive Current Limit

Transfer function $\frac{0.1 \text{ Volt}}{\text{Volt}}$

The gain of this block is zero if any phase voltage or any line-to-line voltage is less than .33 pu. This block acts on the sum of the three phase reactive currents.

Block 11 - Reactive Current Detector

Transfer function $\quad \sin(\theta) \; \dfrac{\text{Volt}}{\text{AMP}}$

Where θ is the angle between phase voltage and current.

Block 12 - Current Limit

Transfer function $\quad \dfrac{.0351 + .07s}{1 + .05s} \; \dfrac{\text{Volt}}{\text{AMP}}$

This block acts on the highest of the phase currents only.

Block 13 - Load Division, Voltage Adjustment

Transfer function $\quad .038 \; \dfrac{1 + .018S}{(1 + .09S)(1 + .0012S)} \; \dfrac{\text{Volts}}{\text{Volt}}$

Block 14 - Phase Discriminator

Locks reference waves together during parallel operation. The input is the phase difference between block 9 and block 9 of paralleled systems.

Transfer function $\quad \dfrac{800\,(1 + .0004S)}{1 + .002S} \; \dfrac{\text{Volt}}{\text{radian}}$

Block 15 - Load Division, Phase Adjustment

Transfer function $\frac{1}{1+S}$ $\frac{\text{Volts}}{\text{Volt}}$

Block 16 - Circulating Current Detector

Detects the part of the current which causes drop in the converter source impedance in phase with the terminal voltage.

Transfer function $\cos(\theta_1 + 38^\circ)$ $\frac{\text{Volt}}{\text{AMP}}$

θ_1 is the angle between the circulating current and the phase voltage.

Block 17 - Circulating Current Detector

Detects the part of the circulating current which causes voltage drop in the converter source impedance at right angles to the terminal voltage.

Transfer function $\sin(\theta_1 + 38^\circ)$ $\frac{\text{Volt}}{\text{AMP}}$

Block 18 - Exciter-Generator

Transfer function for 60, 90 KVA $\frac{17\ N^2}{(1+.02S)^2}$ $\frac{\text{Volts}}{\text{Volt}}$

150 KVA $\frac{34\ N^2}{(1+.02S)^2}$ $\frac{\text{Volt}}{\text{Volt}}$

Where N = generator speed/generator base speed

Block 19 - Generator Regulator Stabalization

Transfer function $\frac{.03}{1 + .04S}$ $\frac{\text{Volts}}{\text{Volt}}$

Block 20 - Amplifier and SCR's

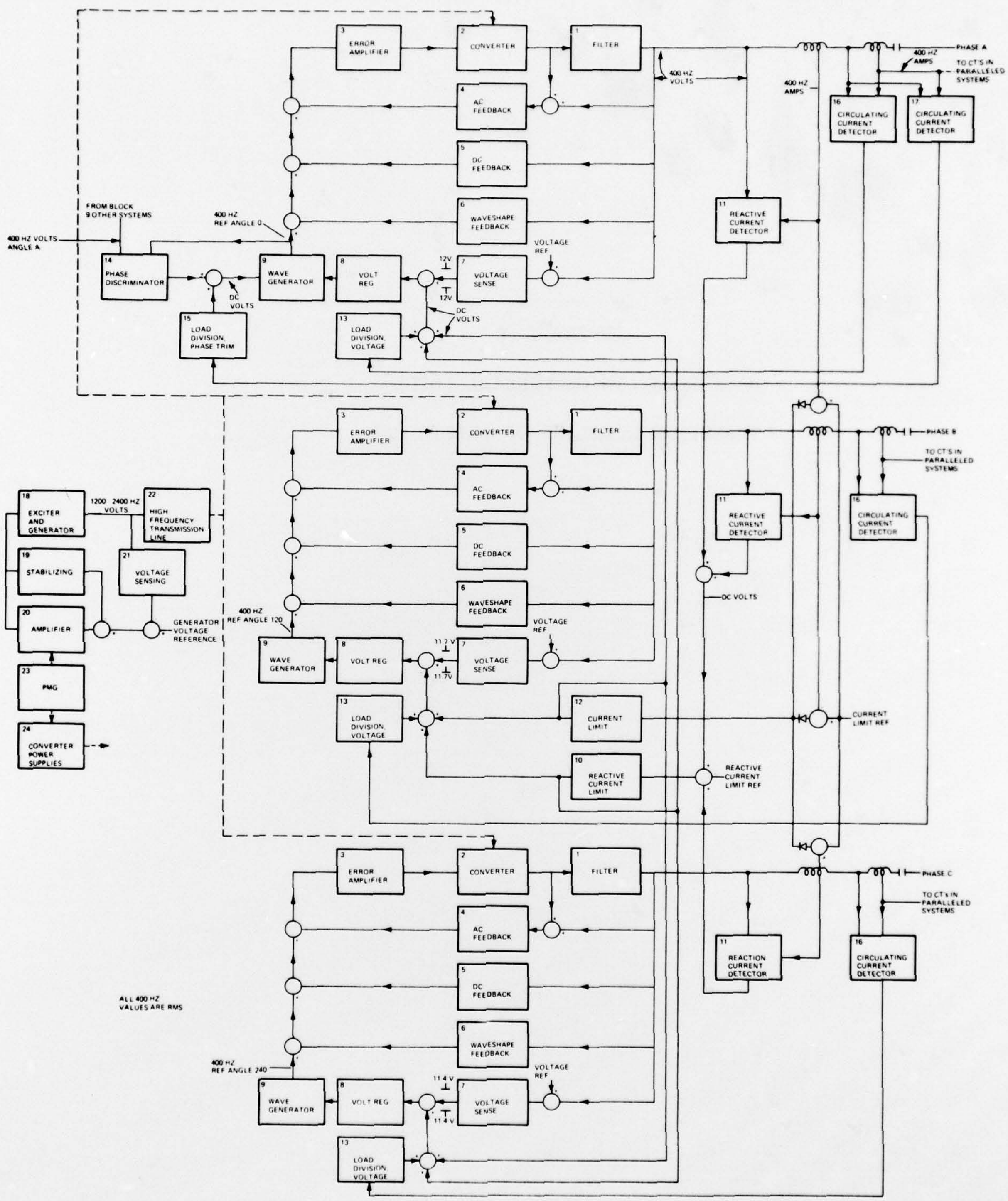
Transfer function $N\,10^5 e^{-st}$ $\frac{\text{Volt}}{\text{Volt}}$

Where t = 1.92/generator RPM seconds

Block 21 - Voltage Sense

Transfer function for 60, 90 KVA $.25\ \frac{\text{Volt}}{\text{Volt}}$

150 KVA $.12\ \frac{\text{Volt}}{\text{Volt}}$



VSCF System Block Diagrams

APPENDIX C

BACKGROUND IN COMMERCIAL (60HZ)

TRANSIENT MODELING OF ELECTRICAL

LOADS

Determining Load Characteristics for Transient Performances
Volume 2: Testing and Modeling of Load Components

EL-849, Volume 2
Research Project 849-3

Final Report, May 1979

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ABSTRACT

This study investigated the dynamic power response of loads (real and reactive) when subjected to excursions of voltage (50% to 120% nominal) and frequency (±5% nominal). Responses of components (air conditioners, motors, lighting, etc.) to changes in voltage and frequency were measured and component load models have been developed from the resulting data. Typical industrial, commercial, and residential loads have been physically simulated and instrumented. A prototype load modeling procedure has been developed.

The period of performance was September, 1976 to June, 1978. The work accomplished by University of Texas at Arlington (UTA) is reported in three volumes, the contents of which are as follows:

- Volume I: Management Summary - Final Report
  An overview and summary of results are presented. Utility industry involvement during the period of research and applications of research results by various utility companies are also reported in this volume.

- Volume II: Testing and Modeling of Load Components - Final Report
  Basic load components are identified and their individual load characteristics defined. Mathematical models for each of the components are included. Comparisons of predicted and actual results on some composite loads are included. The test procedure, setup and instrumentation are described. Test data are stored on magnetic tape. The format of stored data and the method of retrieving it are defined.

- Volume III: Procedure for Modeling Power System Loads - Final Report*
  The modeling philosophy and the prototype procedure for model building is described. Sensitivity analyses of the model building procedure are reported. A computer program was developed for use in load modeling. Its user's manual is included in the appendix of Volume III, separately bound.

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* Volume III is a final report in the context of the RP849-3 contract; however, since its contents are being evaluated under RP849-1, it is considered an interim report in the context of RP849.

LIST OF ABBREVIATIONS

AMUX	Avionics Multiplexing
APU	Auxiliary Power Unit
BC	Bus Controller
BCU	Bus Control Unit
BIT	Built-In-Test
CFG	Constant Frequency Generator
CITS	Central Integrated Test System
CSD	Constant Speed Drive
DAIS	Digital Avionic Information System
EMUX	Electrical Multiplexing
F/O	Fiber Optic
GCU	Generator Control Unit
HVDC	High Voltage Direct Current
IDG	Integrate Drive Generator
ILMC	Integrated Load Management Center
ISD	Integrated Starter Drive
JFS	Jet Fuel Starter
LCC	Life Cycle Cost
LMC	Load Management Center
MUX/DEMUX	Multiplex-Demultiplex (Universal) Terminal
NDRO	Non Destruct Read Only
PDS	Power Distribution System
PMG	Permanent Magnet Generator
RCCB	Remote Control Circuit Breaker
SCU	Standby Control Unit
SSPC	Solid State Power Controller
TSP	Twisted Shielded Pair
VSCF	Variable Speed Constant Frequency
UT	Universal Terminal

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